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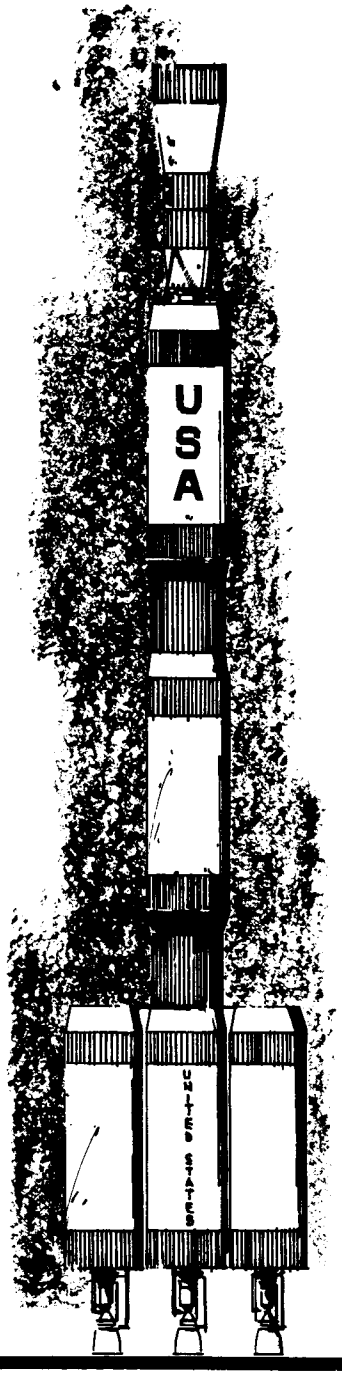
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FACILITY FORM 602



INTEGRATED MANNED INTERPLANETARY SPACECRAFT CONCEPT DEFINITION

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Volume III

D2-113544-3-1

System Analysis (part 1—Missions and Operations)

The BOEING Company • Aerospace Group • Space Division • Seattle, Washington

NASA CR-66560

INTEGRATED MANNED INTERPLANETARY SPACECRAFT CONCEPT DEFINITION

FINAL REPORT

VOLUME III
SYSTEM ANALYSIS

PART 1
MISSIONS AND OPERATIONS

D2-113544-3-1

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

LANGLEY RESEARCH CENTER

Hampton, Virginia

NASA CONTRACT NAS1-6774

January 1968

Distribution of this report is provided in the interest of information exchange. Responsibility for the contents resides in the author or organization that prepared it.

AEROSPACE GROUP THE BOEING COMPANY SEATTLE, WASHINGTON

RECOMMENDED INTERPLANETARY MISSION SYSTEM

The recommended interplanetary mission system:

- Is flexible and versatile
- Can accomplish most of the available Mars and Venus missions
- Is highly tolerant to changes in environment, go-ahead dates, and funding.

It provides:

- Scientific and engineering data acquisition during all mission phases
- Analysis, evaluation, and transmission of data to Earth
- Return to Earth of Martian atmosphere and surface samples

The mission system is centered around the *space vehicle* which consists of the *space acceleration system* and the *spacecraft*.

The *space acceleration system* consists of five identical nuclear propulsion modules:

- Three in the Earth departure stage
- A single module in the planet deceleration stage
- A single module in the planet departure stage

Propellant is transferred between the stages, as necessary, to accommodate the variation in ΔV requirements for the different missions. This arrangement provides considerable discretionary payload capacity which may be used to increase the payload transported into the target planet orbit, the payload returning to the Earth, or both.

The *spacecraft* consists of:

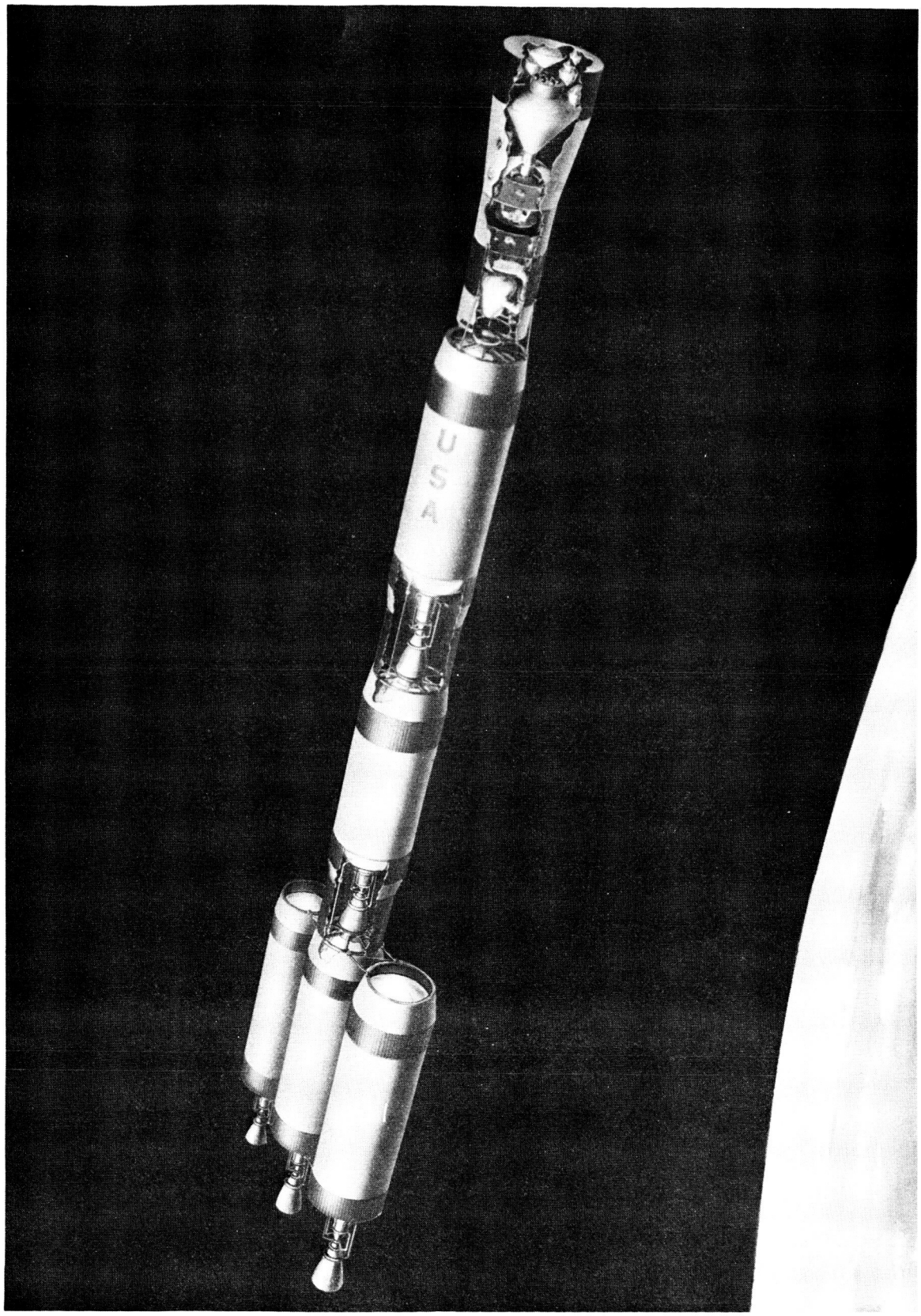
- A biconic Earth entry module capable of entry for the most severe missions
- An Apollo-shaped Mars excursion module capable of transporting three men to the Mars surface for a 30-day exploration and returning
- A mission module which provides the living accommodations, system control, and experiment laboratories for the six-man crew
- Experiment sensors and a planet probe module

The spacecraft and its systems have been designed to accomplish the most severe mission requirements. The meteoroid shielding, expendables, system spares, and mission-peculiar experiment hardware are off-loaded for missions with less stringent requirements.

The space vehicle is placed in Earth orbit by six launches of an uprated Saturn V launch vehicle which has four 156-inch solid rocket motors atttached to the first stage. Orbital assembly crew, supplies and mission crew transportation are accomplished with a six-man vehicle launched by a Saturn IB.

A new launch pad and associated facility modifications are necessary at Launch Complex 39 at Kennedy Space Center to accommodate:

- The weight and length of the uprated Saturn V
- The launch rate necessary for a reasonable Earth orbit assembly schedule
- The solid rocket motors used with the uprated Saturn V
- The requirement for hurricane protection at the launch pad.


USA

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ABSTRACT

In this volume, trajectories are defined for missions to Mars and Venus during the time period 1975 to 1990. Analyses of Earth departure penalties and elliptical orbits at the planets are included. Trajectory optimizations are presented for the case of aerobraking at the planets. Mission profiles are defined and crew workloads are derived. Estimates of mission success probability are given and the abort capability is presented for the recommended IMISCD space vehicle.

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FOREWORD

This study was performed by The Boeing Company for the National Aeronautics and Space Administration, Langley Research Center, under Contract NAS1-6774. The Integrated Manned Interplanetary Spacecraft Concept Definition Study was a 14-month effort to determine whether a variety of manned space missions to Mars and Venus could be accomplished with common flight hardware and to define that hardware and its mission requirements and capabilities. The investigation included analyses and trade studies associated with the entire mission system: the spacecraft; launch vehicle; ground, orbital, and flight systems; operations; utility; experiments; possible development schedules; and estimated costs.

The results discussed in this volume are based on extensive total system trades which can be found in the remaining volumes of this report. Attention is drawn to Volume II which has been especially prepared to serve as a handbook for planners of future manned planetary missions.

The final report is comprised of the following documents, in which the individual elements of the study are discussed as shown:

Volume	Title	Part	Report No.
I	Summary		D2-113544-1
II	System Assessment and Sensitivities		D2-113544-2
III	System Analysis	Part 1--Missions and Operations	D2-113544-3-1
		Part 2--Experiment Program	D2-113544-3-2
IV	System Definition		D2-113544-4
V	Program Plans and Costs		D2-113544-5
VI	Cost-Effective Subsystem Selection and Evolutionary Development		D2-113544-6

The accompanying matrix is a cross-reference of subjects in the various volumes.

● Primary Discussion

X Summary or Supplemental Discussion

STUDY AREAS	DOCUMENTATION					
	Volume I/D2-113544-1 Summary Report	Volume II/D2-113544-2 System Assessment and Sensitivities	Volume III/D2-113544-3 System Analysis	Volume IV/D2-113544-4 System Definition	Volume V/D2-113544-5 Program Plans and Cost	Volume VI/D2-113544-6 Cost Effective Subsystem Selection and Evolutionary Development
			Part 1 - Missions and Operations / Part 2 - Experiment Program			
MISSION ANALYSIS	X		●			
Trajectories and Orbits	X	X	●	X		
Mission and Crew Operations	X	X	●	X		
Mission Success and Crew Safety Analysis	X	X	● ●	X		
Environment	X	X	●			
Scientific Objectives	X	X	●			
Manned Experiment Program	X	X	●			
Experiment Payloads and Requirements	X	X	●			
DESIGN ANALYSIS	X			●		
Space Vehicle	X			●		
Spacecraft Systems	X			●		
Configurations	X	X		●		
Subsystems	X	X		●		
Redundancy and Maintenance	X			●		
Radiation Protection	X			●		
Meteoroid Protection	X			●		
Trades	X			●		
Experiment Accommodations	X			●		
Space Acceleration Systems	X			●		
Primary Propulsion--Nuclear	X	X		●		
Secondary Propulsion--Chemical	X	X		●		
System and Element Weights	X	X		●		
IMIEO Computer Program				●		
Earth Orbit Operations and Assembly Equip.	X	X		●		
Earth Launch Vehicles	X	X		●		
Facilities	X	X		●		
System Trades	X			●		
Space Acceleration--Earth Launch Vehicle	X			●		
Space Acceleration Commonality	X			●		
Space Vehicle--Artificial Gravity	X			●		
SYSTEM AND PROGRAM ASSESSMENT	X	●				
System Capability	X	●	●		X	
Design Sensitivities	X	●		X		
Program Sensitivities	X	●				
Adaptability to Other Space Programs	X	●				
Impact on Other Space Programs	X	●				
Technology Implications	X	●				
Future Sensitivity Studies	X	●				
Program Schedules and Plans	X	X			●	
Test Program	X	X			●	
Facilities Plan	X	X			●	
Program Cost	X	X			●	
Cost Effective Subsystems						●

ABBREVIATIONS

A.U.	Astronomical unit
bps	Bits per second
C/O	Checkout
CM	Command module (Apollo program)
CMG	Control moment gyro
CONJ	Conjunction
CSM	Command service module (Apollo program)
ΔV	Incremental velocity
DSIF	Deep Space Instrumentation Facility
DSN	Deep Space Network
$\oplus$	Earth
ECLS	Environmental control life support system
ECS	Environmental control system
EEM	Earth entry module
ELV	Earth launch vehicle
EMOS	Earth mean orbital speed
EVA	Extravehicular activity
FY	Fiscal year
fps	feet/sec
GSE	Ground support equipment
IBMC	Inbound midcourse correction
IMIEO	Initial mass in Earth orbit
IMISCD	Integrated Manned Interplanetary Spacecraft Concept Definition
I_{sp}	Specific impulse
IU	Instrument unit
KSC	Kennedy Space Center
λ'	Ratio of propellant weight to overall propulsion module weight
LC	Launch complex
LC-34 & -37	Launch complexes for Saturn IB
LC-39	Launch complex for Saturn V
LH_2	Liquid hydrogen
LO	Long
LO_2 or LOX	Liquid oxygen
LRC	Langley Research Center

ABBREVIATIONS (Continued)

LSS	Life support system
LUT	Launch umbilical tower
♂	Mars
MEM	Mars excursion module
MIMIEO	Minimum initial mass in Earth orbit
MM	Mission module
MODAP	Modified Apollo
MSC	Manned Spacecraft Center (Houston)
MSFC	Marshall Space Flight Center (Huntsville)
MTF	Mississippi Test Facility
NAC	Letters designate the type of acceleration systems
	First letter--Earth orbit depart
	Second--planetary deceleration
	Third--planet escape
	Example: NAC = Nuclear Earth depart/aerobraker deceleration at planet/chemical planet escape
OBMC	Outbound midcourse correction
OPP	Opposition
OT	Orbit trim
P/L	Payload
PM-1	Propulsion module, Earth orbit escape
PM-2	Propulsion module, planet braking
PM-3	Propulsion module, planet escape
RCS	Reaction control system
SA	Space acceleration
S/C	Spacecraft
S-IC	First stage of Saturn V
S-II	Second stage of Saturn V
SH	Short
SOA	State of art
SRM	Solid rocket motor
S/V	Space vehicle
SWBY	Swingby

ABBREVIATIONS (Continued)

T/M	Telemetry
TVC	Thrust vector control
VAB	Vehicle assembly building
♀	Venus
V_{HP}	Hyperbolic excess velocity

CONVERSION FACTORS

English to International Units

Physical Quantity	English Units	International Units	Multiply by
Acceleration	ft/sec^2	m/sec^2	3.048×10^{-1}
Area	ft^2	m^2	9.29×10^{-2}
	in^2	m^2	6.45×10^{-4}
Density	lb/ft^3	Kg/m^2	16.02
	lb/in^3	Kg/m^2	2.77×10^4
Energy	Btu	Joule	1.055×10^3
Force	lbf	Newton	4.448
Length	ft	m	3.048×10^{-1}
	n.mi.	m	1.852×10^3
Power	Btu/sec	watt	1.054×10^3
	Btu/min	watt	17.57
	Btu/hr	watt	2.93×10^{-1}
Pressure	Atmosphere	$Newton/m^2$	1.01×10^3
	lbf/in^2	$Newton/m^2$	6.89×10^3
	lbf/ft^2	$Newton/m^2$	47.88
Speed	ft/sec (fps)	m/sec	3.048×10^{-1}
Volume	in^3	m^3	1.64×10^{-5}
	ft^3	m^3	2.83×10^{-2}

CONTENTS

		Page
1.0	INTRODUCTION	1
2.0	MISSIONS	3
	2.1 Mission Matrix	3
	2.1.1 Selected Trajectories	5
	2.1.2 Earth Assembly Orbit	13
	2.1.3 Trajectory Computation and Selection Method	13
	2.2 Supporting Studies	25
	2.2.1 Elliptical Orbits about the Planets	31
	2.2.2 Earth Departure Penalties	34
	2.2.3 Aerobraking Mission Trajectories	46
3.0	OPERATIONS	139
	3.1 Mission Operations	139
	3.1.1 Selected Operational Sequence	139
	3.1.2 Function and Task Analysis	143
	3.2 Support Operations	143
	3.2.1 Orbital Support	143
	3.2.2 Earth-Based Support Operations	171
	3.3 Crew Operations	174
	3.3.1 Work Loads	174
	3.3.2 Crew Skills	178
	3.3.3 Vehicle Operations	178
	3.3.4 Vehicle Maintenance	182
	3.3.5 Experimentation and Exploration	183
	3.3.6 Crew Care	188
4.0	MISSION SUCCESS AND CREW SAFETY ANALYSIS	195
	4.1 Mission Success Probability	195
	4.1.1 Mission Module Reliability	195
	4.1.2 Reliability Allocation	196
	4.1.3 Single-Vehicle Mode Mission Success Probability	196
	4.1.4 Convoy Mode--OVAM	198
	4.1.5 Convoy Mode (S+C)	204
	4.1.6 Convoy Mode Comparisons	204
	4.2 Abort Analysis	223

CONTENTS (Continued)

		Page
5.0	MISSION ENVIRONMENTS	233
5.1	Earth Surface Environment	233
5.1.1	Temperature	233
5.1.2	Ambient Pressure	233
5.1.3	Humidity	233
5.1.4	Solar Radiation	233
5.1.5	Atmosphere	234
5.2	Earth Orbital Environment	234
5.2.1	Ambient Pressure	234
5.2.2	Terrestrial Infrared Radiation	234
5.2.3	Solar Radiation	234
5.2.4	Reflected Solar Radiation	234
5.2.5	Charged Particle Radiation	234
5.2.6	Meteoroid Flux	235
5.3	Interplanetary Environment	237
5.3.1	Ambient Pressure	237
5.3.2	Solar Radiation	237
5.3.3	Charged Particle Radiation	237
5.3.4	Meteoroid	241
5.4	Mars Environment	241
5.4.1	Model Atmospheres (NASA)	241
5.4.2	Planet Solar Radiation	250
5.4.3	Charged Particle Radiation	250
5.4.4	Meteoroid	251
5.5	Venus Environment	251
5.5.1	Model Atmospheres	251
5.5.2	Planet Solar Radiation	251
5.5.3	Charged Particle Radiation	251
5.5.4	Meteoroid	255

1.0 INTRODUCTION

This volume presents requirements imposed on the IMISCD system design. These include studies on the trajectories, operational sequences and environments, and an assessment of mission success and crew safety with attendant reliability requirements. The mission payloads and other implications of the scientific and engineering mission are contained in Part 2 of Volume III.

Trajectories were generated for the standard classes of missions over the 1975 to 1990 time period. These include Mars opposition, conjunction, and Venus swingby missions and long and short stay-time missions to Venus. The trajectory information provided energy requirements, mission times, and other important design parameters such as spacecraft-to-sun distances and Earth-to-spacecraft communication distances as reported in Section 2.1. Additional supporting analyses of Earth departure penalties and eccentric orbits at the planets indicate a need for further mission analysis to better define total energy requirements for launch windows at the Earth and planets and to correlate these with acceptable planet orbits from the scientific point of view. These studies are reported in Section 2.2.

Operational sequences identify design requirements and provide important inputs to the program planning and costing studies reported in Volume IV. Analysis of these operations provided an assessment of crew requirements in the planetary mission phases. These are contained in Section 3.0. The analysis shows that for a six-man crew the available manhours assignable to the scientific mission is critical from early encounter phases through departure. During the Mars surface operations, an accurate assessment of the crew size requirement is dependent on a more detailed definition of task requirements for the specific scientific apparatus used.

Section 4.0 presents an assessment of mission success and crew safety as well as reliability goals for design. Although substantial abort capability exists in terms of ΔV to return throughout the mission, abort opportunities with short return times are confined to approximately the first 50 days after Earth departure. Comparison of a convoy mode showed no advantage, on the basis of mission success or crew safety, over the single-vehicle mode.

Section 5.0 presents the environmental data used in the design studies.

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2.0 MISSIONS

A major objective of the study was to design a system that could respond to the majority of the opportunities for trips to Mars and Venus over the time period 1975 through 1990. To support the system trade and design studies, a matrix of missions was developed to cover both the variations due to date and the variations due to class of trajectories. Although the use of elliptical orbits at the planets affords an opportunity to reduce mission energy requirements and to smooth out the variation in energy requirements over the range of opportunities, it was decided that the objectives of the study were better served by fixing the planet orbits as circular at 1000 kilometer. From a scientific standpoint, circular orbits are more desirable than elliptical orbits, and the resulting system design remains flexible as to alternative planet orbits. The following sections present the selected trajectories, the Earth assembly orbit selection, and trajectory computation and selection methods.

2.1 MISSION MATRIX

Mars and Venus opportunities during the 1975-1990 Mars synodic cycle are shown in Figure 2.1-1 by the solid lines. Those opportunities selected to establish the range of mission requirements for the study are indicated by the symbolic representation of the planets involved. Placement of the symbols grossly indicate Earth departure, target planet arrival, stay and departure, and Earth arrival.

The Mars missions are divided into three classes:

1) Opposition class missions result in round-trip times of approximately 450 to 500 days but require relatively high energy. Planet stay time was fixed at 40 days. For the unfavorable time between 1975 and 1980, the energy requirement is prohibitive for this class of mission, and this condition repeats during each synodic cycle.

2) Venus swingby missions provide trajectories having practical energy requirements during these unfavorable periods, and the mission times are increased only to approximately 550 to 600 days. Again the planet stay time was fixed at 40 days.

3) The Mars conjunction missions require relatively low energy but have durations of about 1000 days with 400 to 600 days planet stay time.

The Venus orbit missions are divided into short-stay and long-stay classes. The long-stay-time mission has the lowest energy requirement but requires stay times of about 450 days, giving total trip times of about 800 days. Short-stay-time missions assume 40 days at Venus. Higher energies are required, but trip times are reduced to about 450 to 500 days. The selected missions cover a Venus synodic cycle of approximately 7 years. Figure 2.1-1 shows a total of 24 missions as follows:

- Four opposition class missions to Mars from 1982 through 1988 (with a 40-day stay time).

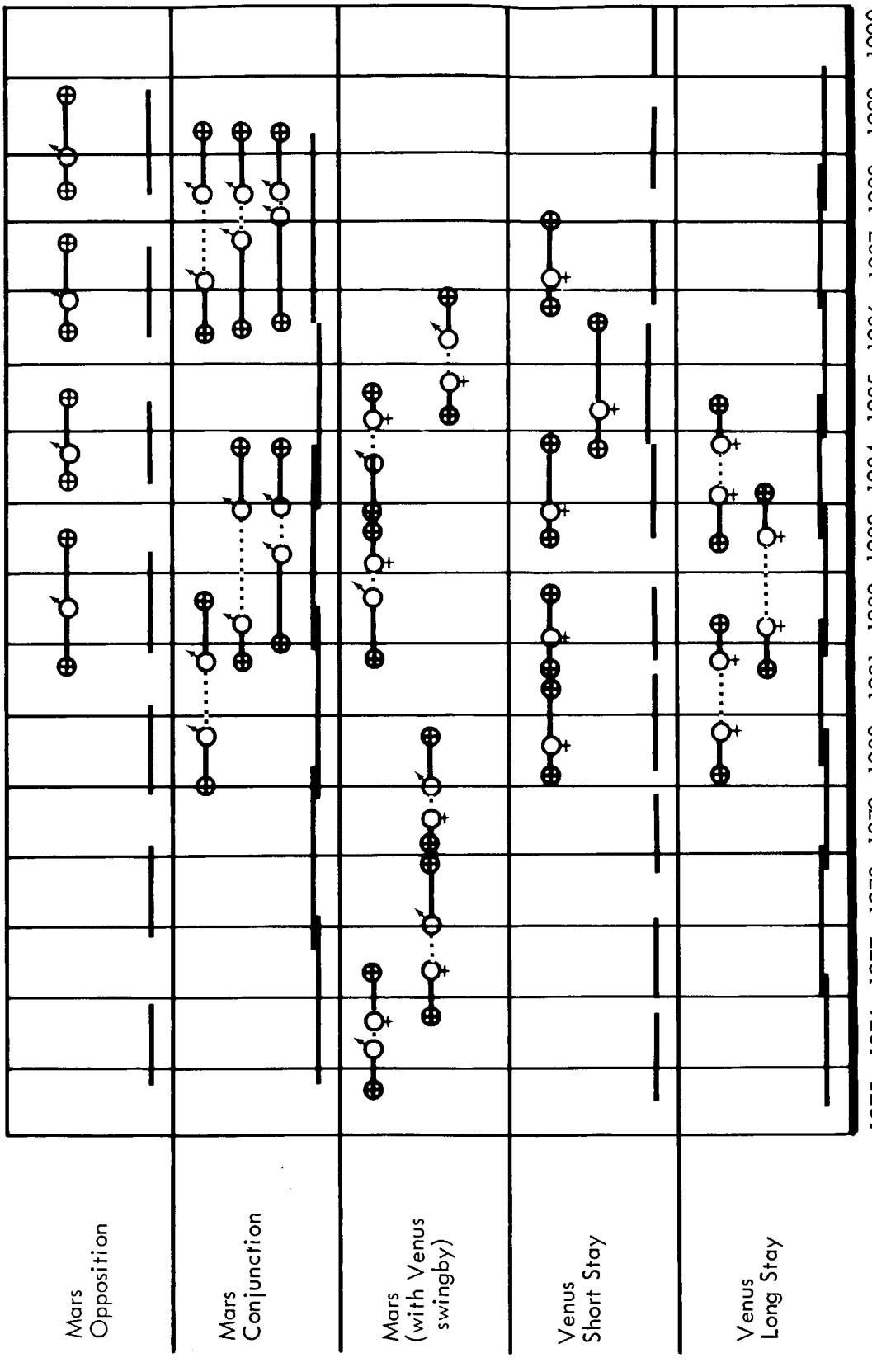


Figure 2.1-1: MARS-VENUS MISSION OPPORTUNITIES (1975 - 1990)

- Two conjunction class missions to Mars to represent the upper and lower bounds on energy requirements for conjunction class missions throughout the time period of interest (1975-1990).
- Two conjunction class missions with long trip times to Mars in order to lengthen the available launch period at Earth.
- Five short-stay-time (40 days), capture missions to Venus. Because of the close repeatability of conditions, these five represent all short-stay-time missions from 1975 to 1990.
- Three long-stay-time, capture missions to Venus which similarly represent all long-stay-time missions from 1975 to 1990.
- Three representative, outbound, Venus swingby missions to Mars.
- Three representative, inbound, Venus swingby missions to Mars.

Typical trajectories are shown in Figure 2.1-2.

2.1.1 SELECTED TRAJECTORIES

For each of the selected missions, specific trajectories were defined to provide specific velocity requirements, trip times, and other parameters such as communications distance and Sun distance that could be used in the system trade and design studies. Trajectories were selected on the basis of minimum initial mass in Earth orbit as described in detail in Section 2.1.3. The characteristics of the trajectories thus selected are tabulated in Table 2.1-1. The velocity requirements shown are impulsive, and corrections were made for gravity losses in the design studies based on specific configuration thrust-to-weight ratios. The IMIEO values given in Table 2.1-1 are for reference only; trade study and system design IMIEO values are contained in Volume IV.

A typical detailed trajectory is plotted in Figure 2.1-3 illustrating the distance relationships tabulated in Table 2.1-1. The occultations of the spacecraft from the Earth and the Mars lander are listed in Table 2.1-2.

Table 2.1-2: MAXIMUM OCCULTATIONS OF SPACE VEHICLE FROM EARTH AND MEM

Mission	Occultation Time		
	In Orbit Around Planet		In Heliocentric Orbit - Occult. from Earth
	Occult. from Earth	Occult. from MEM	
Mars Opposition	42 min/orbit	117 min/orbit	68 days
Mars Conjunction	68 days max	117 min/orbit	68 days
Mars-Venus Swingby	42 min/orbit	117 min/orbit	68 days
Venus - Short	33 min/orbit		78 days
Venus - Long	78 days max		78 days

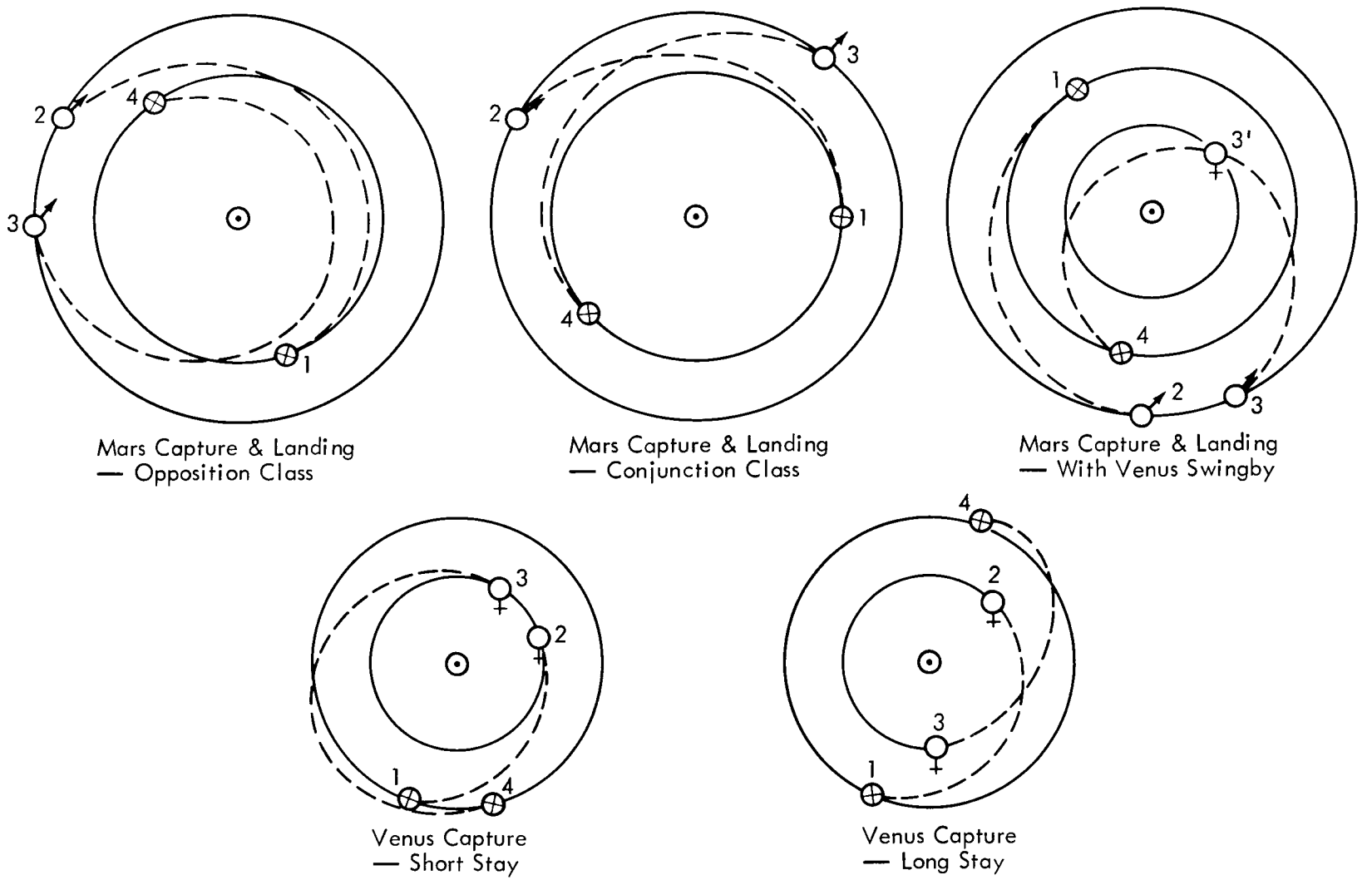


Figure 2.1-2: MISSION TRAJECTORIES (1975-1990 Mars Synodic Cycle)

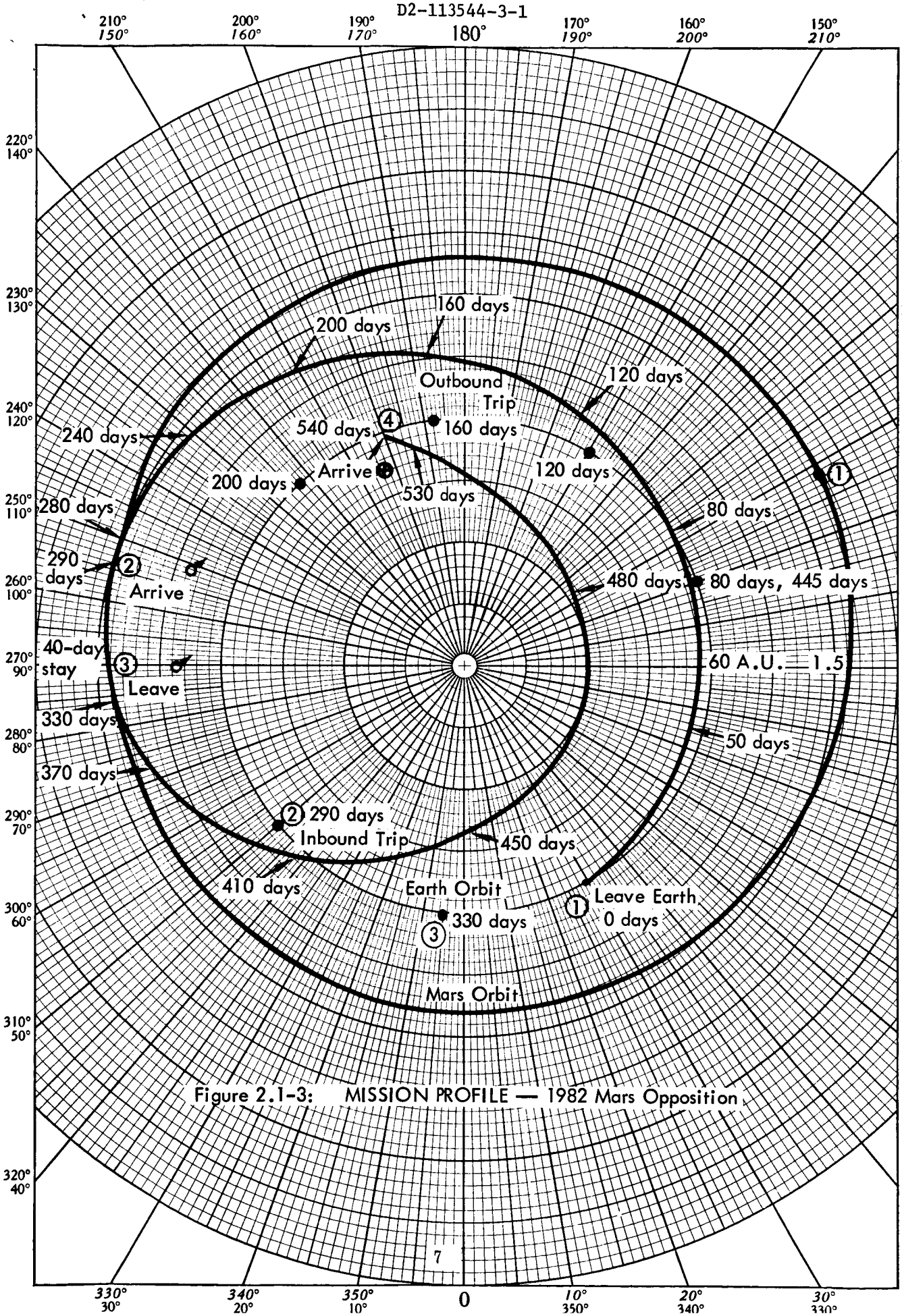


Figure 2.1-3: MISSION PROFILE — 1982 Mars Opposition

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Table 2.1-1 IMISCD MISSIONS

MISSION	ΔV_1 m/sec	ΔV_2 m/sec	ΔV_3 m/sec	TOTAL ΔV m/sec	V_E m/sec	ΔV_4 m/sec	t_1 days	t_2 days	t_3 days	TOTAL TRIP days	V_{HP_1} m/sec	V_{HP_2} m/sec	V_{HP_3} m/sec	V_{HP_4} m/sec	T_1 (244-)	T_2 (244-)	T_3 (244-)	T_4 (244-)	R_{max} (AU)	R_{min} (AU)	D_e (AU)	IMIEO ESTIMATE (10^6 kg)
♂OPP 82	3989	2568	5811	12368	18350	4637	290	40	210	540	4340	3589	7765	14635	4900	5190	5230	5440	1.6	0.502	1.56	0.5040
♂OPP 84	3685	3086	5352	12123	16580	2865	190	40	230	460	3450	4364	7232	12340	5740	5950	5990	6220	1.6	0.569	1.26	0.4207
♂OPP 84(ALT)	3937	2611	5351	11899	16580	2865	280	40	230	550	4201	3657	7232	12340	5670	5950	5990	6220	1.6	0.569	1.26	0.4032
♂OPP 86	3645	2832	4969	11446	16260	2549	190	40	250	480	3322	3995	6779	11913	6520	6710	6750	7000	1.66	0.611	0.99	0.3703
♂OPP 88	3947	2945	5142	12034	17620	3904	170	40	250	460	4228	4161	6985	13704	7320	7490	7530	7780	1.50	1.0	0.999	0.4505
SWBY 75	4379	5312	3701	13392	12664	197 / 0	150	40	190 / 180	560	5294	7185	5202	12285 / 6142	2670	2820	2860	3050 / 3230	1.65	0.511	1.45	0.5604
SWBY 78	5093 / 156	3800	3750	12643	16220	2507	235 / 165	40	270	710	6756 / 7037	5331	5267	11855	3080 / 3315	3480	3520	3790	1.63	0.667	0.855	0.5126
SWBY 80	4900 / 60	4275	2504	11679	13016	0	140 / 140	40	300	620	6386 / 10446	5934	3487	6839	3860 / 4000	4140	4180	4480	1.61	0.555	1.7	0.4221
SWBY 82	3798	2337	4550	10685	12160	106 / 0	260	40	140 / 160	600	3802	3210	6274	11010 / 5021	4910	5210	5250	5390 / 5550	1.46	0.535	1.66	0.3126
SWBY 84	5053	3232	4418	12703	11646	168 / 0	280	40	160 / 160	640	6680	4570	6112	6195 / 3609	5620	5900	5940	6100 / 6260	1.67	0.719	0.977	0.4511
SWBY 86	4316 / 190	3779	2633	10728	12376	0	150 / 200	40	200	590	5151 / 9520	5303	3692	5524	6170 / 6320	6520	6560	6760	1.62	0.588	1.35	0.3533
♂CONJ 80	3869	2124	1926	7919	12015	0	290	370	340	1000	4008	2833	2449	4660	4200	4490	4860	5200	1.67	0.984	2.6	0.2904
♂CONJ 86	3684	2470	2713	8867	11852	0	210	580	250	1040	3447	3432	3815	4222	6530	6740	7320	7570	1.67	0.949	2.674	0.2419
♀SH 80	3858	4538	4070	12466	14190	477	130	40	290	460	3979	6019	5086	8878	4310	4440	4480	4770	1.0	0.722	0.735	0.4270
♀SH 81	3849	4190	4072	12111	14280	567	120	40	300	460	3952	5338	5092	9019	4900	5020	5060	5360	1.0	0.723	0.672	0.3992
♀SH 83	3543	3606	4310	11459	14790	1073	180	40	330	540	2950	4006	5577	9801	5470	5650	5690	6020	1.0	0.715	1.03	0.3577
♀SH 85	3900	3337	4089	11326	14700	980	190	40	320	550	4098	3245	5126	9660	6020	6210	6250	6570	1.0	0.726	0.866	0.3491
♀SH 86	3599	4214	4104	11917	14270	550	130	40	300	470	3158	5387	5159	8994	6650	6780	6820	7120	1.0	0.715	0.739	0.3862
♀LO 80	3661	4539	3400	11600	11751	0	190	430	180	800	3370	6021	3030	3927	4320	4510	4940	5120	1.0	0.723	1.72	0.4218
♀LO 81	3539	3788	3306	10633	11623	0	170	430	170	770	2937	4454	3145	3505	4900	5070	5500	5670	1.0	0.709	1.72	0.3582
♀LO 83	3622	3627	3317	10560	11605	0	190	470	120	780	3235	4057	3184	3469	5450	5640	6110	6230	1.0	0.714	1.72	0.3524
♂LOOT 82	3749	5148	1874	10771	12321	0	390	270	320	980	3653	6992	2340	5399	4990	5380	5650	5970	1.67	0.975	2.21	0.4136
♂LOOT 82	3947	6132	1874	11953	12321	0	430	160	320	910	4226	8132	2340	5399	5060	5490	5650	5970	1.67	0.975	2.54	0.5284
♂LOOT 86	4040	3488	2713	10241	11852	0	420	280	250	950	4475	4919	3815	4222	6620	7040	7320	7570	1.02	0.72	2.674	0.3420
♂LOOT 86	4265	4302	2713	11280	11852	0	550	100	250	900	5031	5967	3815	4222	6670	7220	7320	7570	1.0	0.72	1.656	0.4124

Table 2.1-1-A

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Nomenclature for Table 2.1-1

D_e Maximum distance from Earth to spacecraft while spacecraft is at the target planet.

R_{max} Maximum distance from spacecraft to Sun occurring during the mission.

R_{min} Minimum distance from spacecraft to Sun occurring during the mission.

t_1 Trip time to the target planet. (In the outbound swingby missions, the legs before and after Venus swingby are shown.)

t_2 Stay time at the target planet.

t_3 Trip time from the target planet to Earth. (In the inbound swingby missions, the legs before and after Venus swingby are shown.)

T_1 Julian date of departing Earth (and Julian date of outbound flyby). Modulus - 244-0000

T_2 Julian date of arrival at target planet. Modulus 244-0000

T_3 Julian date of departing target planet. Modulus 244-0000

T_4 Julian date of arriving at Earth (and Julian date of inbound swingby). Modulus - 244-0000

V_{HP_1} Hyperbolic excess velocity at departure (and at outbound swingby).

V_{HP_2} Hyperbolic excess velocity at arrival to target planet.

V_{HP_3} Hyperbolic excess velocity at departure from target planet.

V_{HP_4} Hyperbolic excess velocity at arrival to Earth (and at inbound swingby).

V_E Entry velocity at 65.8-kilometer altitude at Earth.

ΔV_1 Velocity increment required from a 537-kilometer circular orbit at Earth (and trimming velocity increment required at outbound swingby for this specific trajectory).

ΔV_2 Velocity increment required to enter circular orbit at the target planet.

ΔV_3 Velocity increment required to depart from circular orbit at the target planet.

ΔV_4 Velocity increment required at Earth to reduce entry velocity at 122-kilometer altitude to 13.7 km/sec.

Mission Abbreviations for Table 2.1-1

OPP	Opposition mission (40-day stay time at Mars)
CONJ	Conjunction mission
SH	Short stay-time mission (40 day stay time at Venus)
LO	Long stay-time mission
SWBY	Venus swingby mission to Mars with nearly zero ΔV applied at Venus
LOOT	Long outbound trip to Mars (to extend launch period)

2.1.2 EARTH ASSEMBLY ORBIT

One of the primary criteria for orbit selection from the standpoint of tracking, communications, and Earth-based support, in general, is that of having the same ground track each day; i.e., the orbiting vehicle will be in the same position relative to some point on Earth each day. This is particularly true for missions that require orbital assembly operations lasting more than a day or so. Considering the orbital altitude and period versus the Earth's rotation, plus the second order effects due to the Earth's oblateness, orbits can be determined that will satisfy this criteria. However, those orbits that qualify under this criteria are defined by specific altitudes and inclinations. The curves of Figure 2.1-4 show the relationship between altitude and orbit inclination for these repeating orbits. Also shown are the limits of orbital inclination (28.5 to 34.3 degrees) set by present launch azimuth restrictions of 70 to 110 degrees and desirable altitude limits. The latter are a lower limit due to atmospheric drag and an upper limit due to radiation, particularly in the South Atlantic anomaly. Since the declinations of the Earth departure asymptotes for the selected missions are generally below 28.5 degrees, the familiar 28.5 degree/262 nautical-miles (485 km) repeating orbit was chosen for the study. A more detailed consideration of departure asymptote variations in regard to launch window as discussed in Section 2.2.2 might suggest the selection of different inclinations for specific missions.

2.1.3 TRAJECTORY COMPUTATION AND SELECTION METHOD

A single, multiplanet computer program was used to compute the trajectories. The program matches a series of two-body, conic, trajectory legs into one complete trajectory. It accepts as inputs the Julian date departing Earth, the Julian date arriving at each subsequent planet, and the stay time at each planet. Additional inputs of propulsion system specific impulse, propulsion stage mass fraction, orbit sizes, masses of the various vehicle modules, and the expended weight loss per day are used to compute the required vehicle mass at each terminal point of the interplanetary trajectories.

The procedure for trajectory selection employed the results of previous studies of minimum required initial mass in Earth orbit (MIMIEO) to locate the most favorable regions of study for the IMISCD missions. Around the published MIMIEO and minimum total ΔV round-trip trajectories to Mars, a permutation grid in 10-day increments of departure and arrival dates was used. Estimates of vehicle weights and propulsion system characteristics were used to calculate the necessary initial mass in Earth orbit (IMIEO) for each trajectory. These are given in Table 2.1-3. The missions to Venus were analyzed in the same manner, except that the initial range of interest for the departure and arrival dates was obtained by inspection of contour plots in the NASA planetary flight handbook,* and the permutations of dates were in 20-day increments.

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*NASA SP-35 Series, Space Flight Handbooks, *Planetary Flight*, Vol. 3.

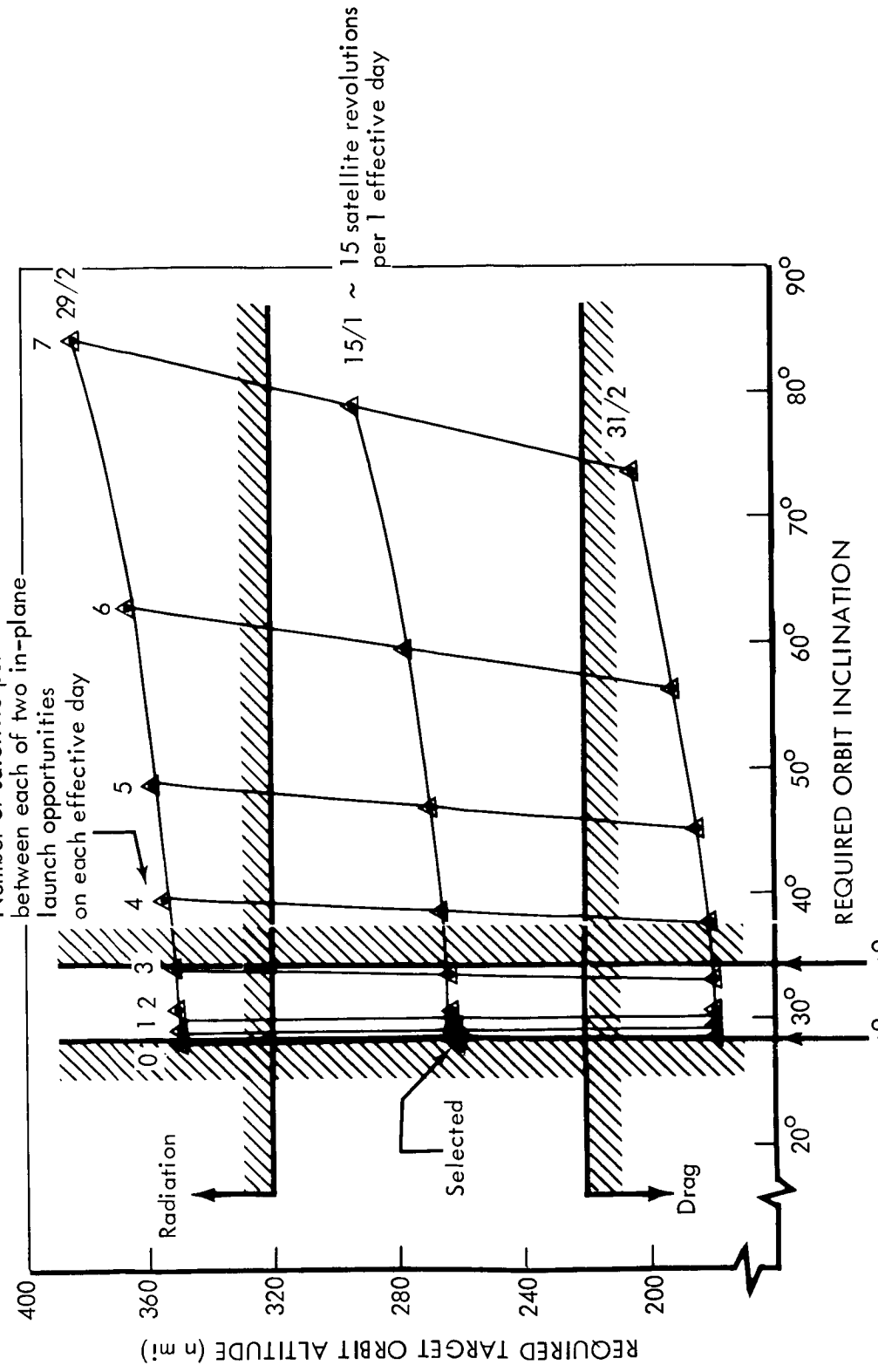


Figure 2.1-4: REQUIRED SYNCHRONOUS ORBIT ALTITUDE AND INCLINATION FOR IN-PLANE/IN-PHASE LAUNCH OPPORTUNITIES

Table 2.1-3: VEHICLE CHARACTERISTICS

Vehicle Characteristic	Missions				
	Mars Conjunction	Mars Opposition	Mars Venus Swingby	Venus Long	Venus Short
Earth Entry Module Weight (lb)	15,000	15,000	15,000	15,000	15,000
Stage 4 Specific Impulse (sec) I_{sp}	410	410	410	410	410
Stage 4 Mass Fraction, (λ')	0.945	0.945	0.945	0.945	0.945
Basic Mission Module Weight (lb)	87,000	73,400	*See table below	89,000	73,400
Expended Mass Flow (lb/day loss)	10	10	10	10	10
Stage 3 I_{sp} (sec)	850	850	850	850	850
Stage 3 (λ')	0.838	0.824	0.824	0.838	0.824
MEM Weight (lb)	55,000	55,000	55,000	55,000	55,000
Stage 2 I_{sp} (sec)	850	850	850	850	850
Stage 2 (λ')	0.767	0.759	0.759	0.767	0.759
Stage 1 I_{sp} (sec)	850	850	850	850	850
Stage 1 (λ')	0.816	0.823	0.823	0.816	0.823

Date	1975	1978	1980	1982	1984	1986
*Module weights for Mars-Venus swingby (lb)	80,800	79,000	80,600	79,800	80,000	80,000

For the Venus swingby missions to Mars, the initial trajectory estimate was obtained from published data, and the permutation increments were 20 days at Earth and Mars, with 5-day (outbound) and 10-day (inbound) increments at Venus.

An arbitrary allowance was made for Earth orbit departure launch window in the selection of specific trajectories. The procedure used was to select a trajectory near minimum IMIEO, but at an IMIEO level that is not exceeded over at least a 30-day period. This generally resulted in IMIEO's less than 10% above the minimum.* Complete treatment of the launch window problem requires a more sophisticated analysis which was not warranted for this study. However, some additional aspects were investigated and are discussed in Sections 2.2.2 and 2.2.3. No allowance was made for launch window at the planets.

2.1.3.1 Opposition Class Mars Missions and Short Missions to Venus

Because both the outbound and inbound legs of these missions are compromised to reduce trip time, each round trip trajectory is compared as an entity. Figure 2.1-5 for the 1982 Mars opposition mission is exemplary of the data reduction techniques used. First, all trajectories were plotted for IMIEO as a function of arrival date at Earth in lines of constant arrival date at Mars and constant Earth departure date. These lines separate into families of lines and each family represents a single arrival date at Mars. In the actual flight, the arrival date at any planet can be adjusted accurately through midcourse corrections. Therefore, the complete trajectory was selected by crossplotting the minimums of all the previously mentioned lines into a plot of IMIEO versus Earth departure date at constant Mars arrival date. This latter plot contains all the information required for trajectory selection. In the case of Figure 2.1-4, the selected nominal mission is launched 20 days before the minimum IMIEO trajectory. A check of the graph shows that the selected trajectory has less than 10% IMIEO penalty over the minimum IMIEO trajectory. The remainder of the Mars opposition missions and the Venus short missions are analyzed in the same manner. Graphs for these trajectories are shown in Appendix A-1 which follows this section.

2.1.3.2 Heliocentric Plane Change

A portion of Figure 2.1-5's outbound leg plot shows a region where the transfer takes nearly 180 degrees travel around the Sun. This condition shows a large increase in IMIEO at Earth departure dates near Julian date 244-4950. To reduce this peak, a set of trajectories was run in

*The exception to this is the 1980 Venus swingby, where the grid size results in trajectories with an IMIEO approximately 125% of MIMIEO and a 35-day launch period.

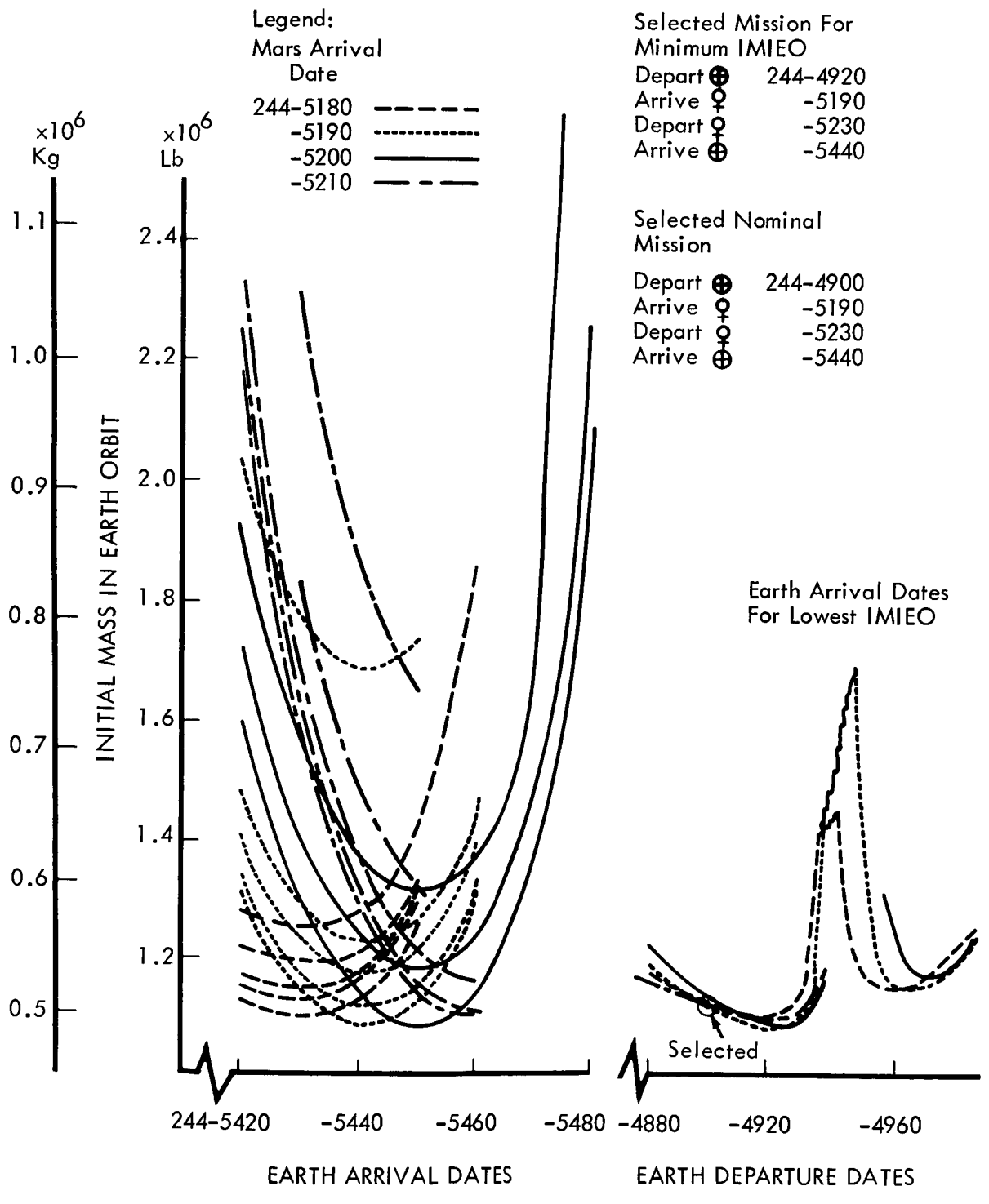


Figure 2.1-5: 1982 MARS OPPOSITION-CLASS TRAJECTORIES

1982 where a large midcourse correction for heliocentric plane change was allowed. This allows trajectories nearer to the ecliptic and reduces the IMIEO required near the 180-degree transfer point. Nevertheless, as shown in Figure 2.1-15, this does not provide a significant advantage when only a 30-day launch period is required. Similar results for 1986 imply that plane change is of little advantage in the entire mission range, and trajectories of this type were not used.

2.1.3.3 Mars Conjunction Class Missions and Venus Long Missions

Because the expended weight flow from the spacecraft is low, the outbound leg and inbound leg of the trajectory were initially optimized separately. This initial graphical selection was subsequently checked from the printout, considering the 10 pounds/day weight loss, to ascertain that the proper trajectory was selected. Figure 2.1-16 is an example of the data graph for these trajectories. For convenience, the inbound leg is described by the required mass departing Mars plotted against Earth arrival date, for lines of constant Mars departure date. The outbound leg is plotted as the ratio of mass departing Earth and mass placed in orbit about Mars. This ratio is relatively insensitive to variations in the inbound trajectory leg. In most cases, total trajectories using the optimum inbound leg were used to provide the data plotted for the outbound leg; however, it was found that this was not a critical requirement. Appendix A-2 contains the curves used to select trajectories for the long Venus missions and conjunction class Mars missions.

Discounting departure penalties due to orbit precession and day-by-day variations in declination of the departure asymptote, it is possible, by allowing a longer outbound trip, to extend the available launch period at Earth. Figures 2.1-21 and 2.1-22 show these trajectories for the 1982 and 1986 Mars missions. Figure 2.1-23 shows the IMIEO penalty involved in this means of extending the launch period. This figure is a combination of data derived from Figures 2.1-16 and 2.1-22. A 160-day launch period is available for approximately a 50% increase in IMIEO, considering trajectory energy requirements only. For a 10% increase in IMIEO, an 80-day launch window is available. A more sophisticated analysis would be required to determine the usefulness of this feature of the conjunction class missions.

2.1.3.4 Venus Swingby Missions

In matching the approach and departure trajectories at Venus swingby, the passage distance that provided the proper amount of turning was used. If the approach and required departure speeds differed, the difference was provided by propulsion after swingby. If the turning required a passage below 200-kilometer altitude, then the additional turning was performed with propulsion. Manual scanning of the data allowed selection and direct plotting of those trajectories that require minimal propulsion. The machine program printout was manually scanned and the trajectories with low ΔV requirements at Venus were tabulated. Then, at each Earth departure date, the trajectory that required the least ΔV at Venus (specifically less than 200 m/sec) and the least IMIEO was plotted as in Figure 2.1-29. These trajectories are referred to as unpowered.

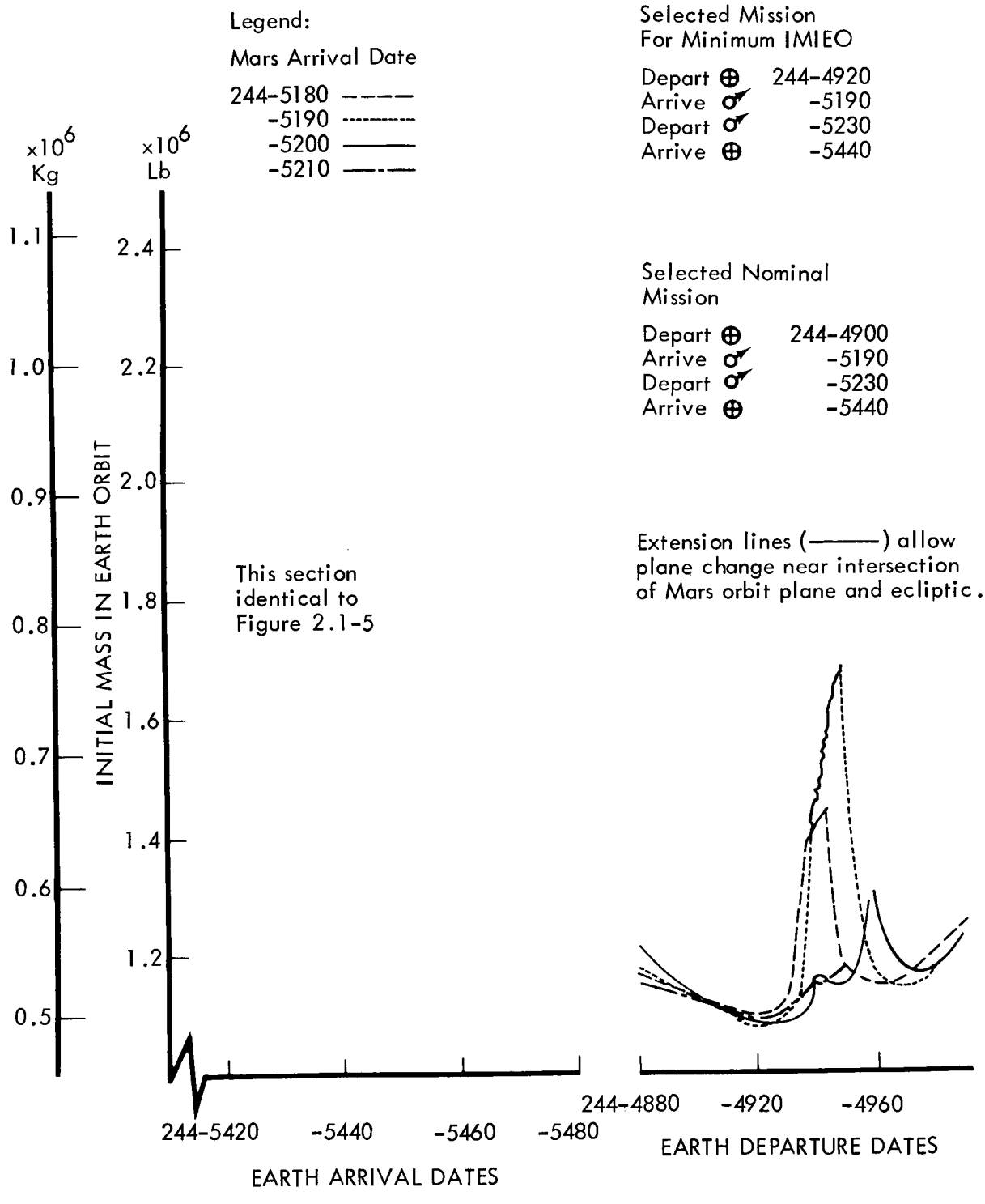


Figure 2.1-15: 1982 PLANE CHANGE MARS OPPOSITION-CLASS TRAJECTORIES

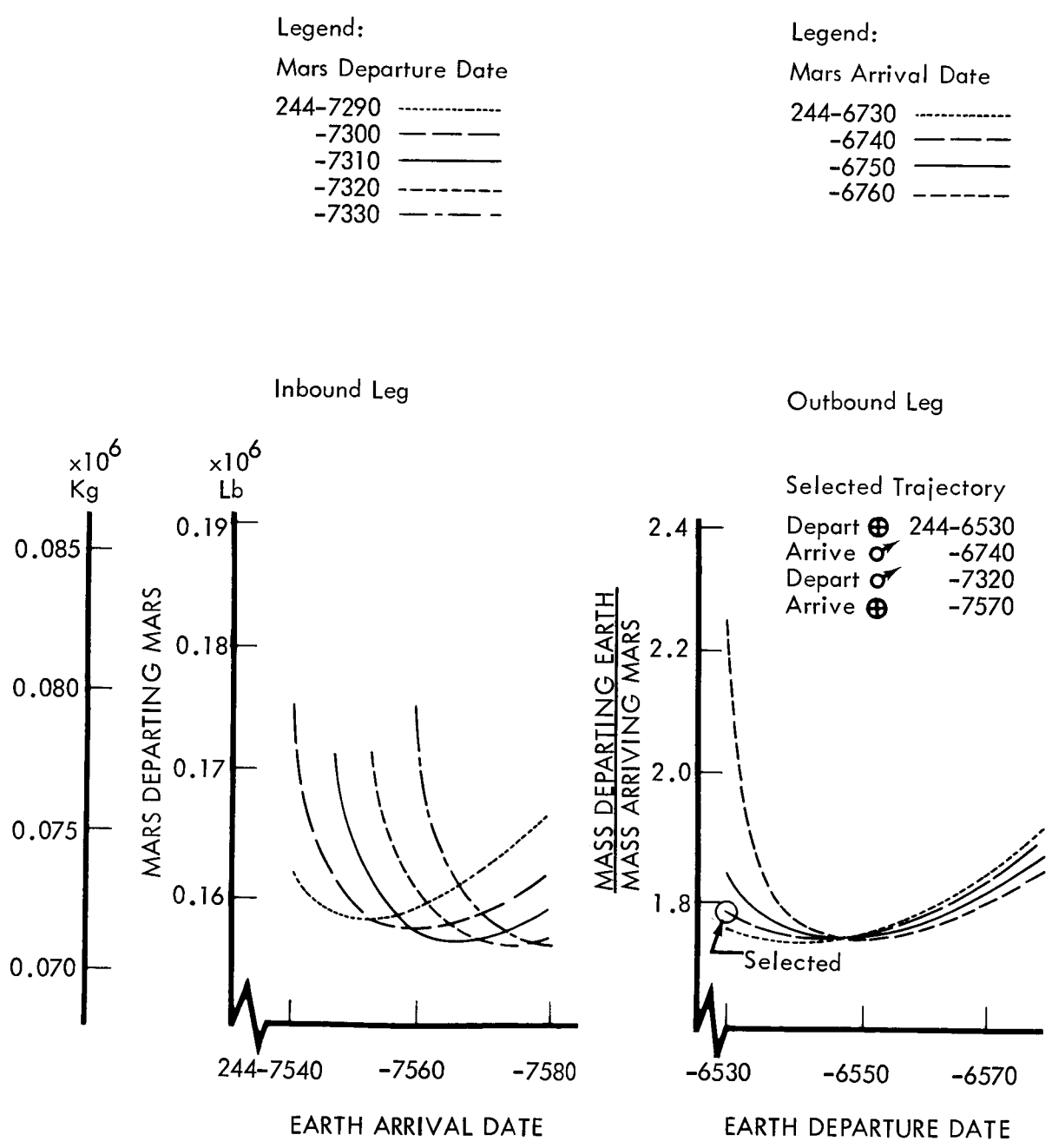


Figure 2.1-16: 1986 MARS CONJUNCTION-CLASS TRAJECTORIES

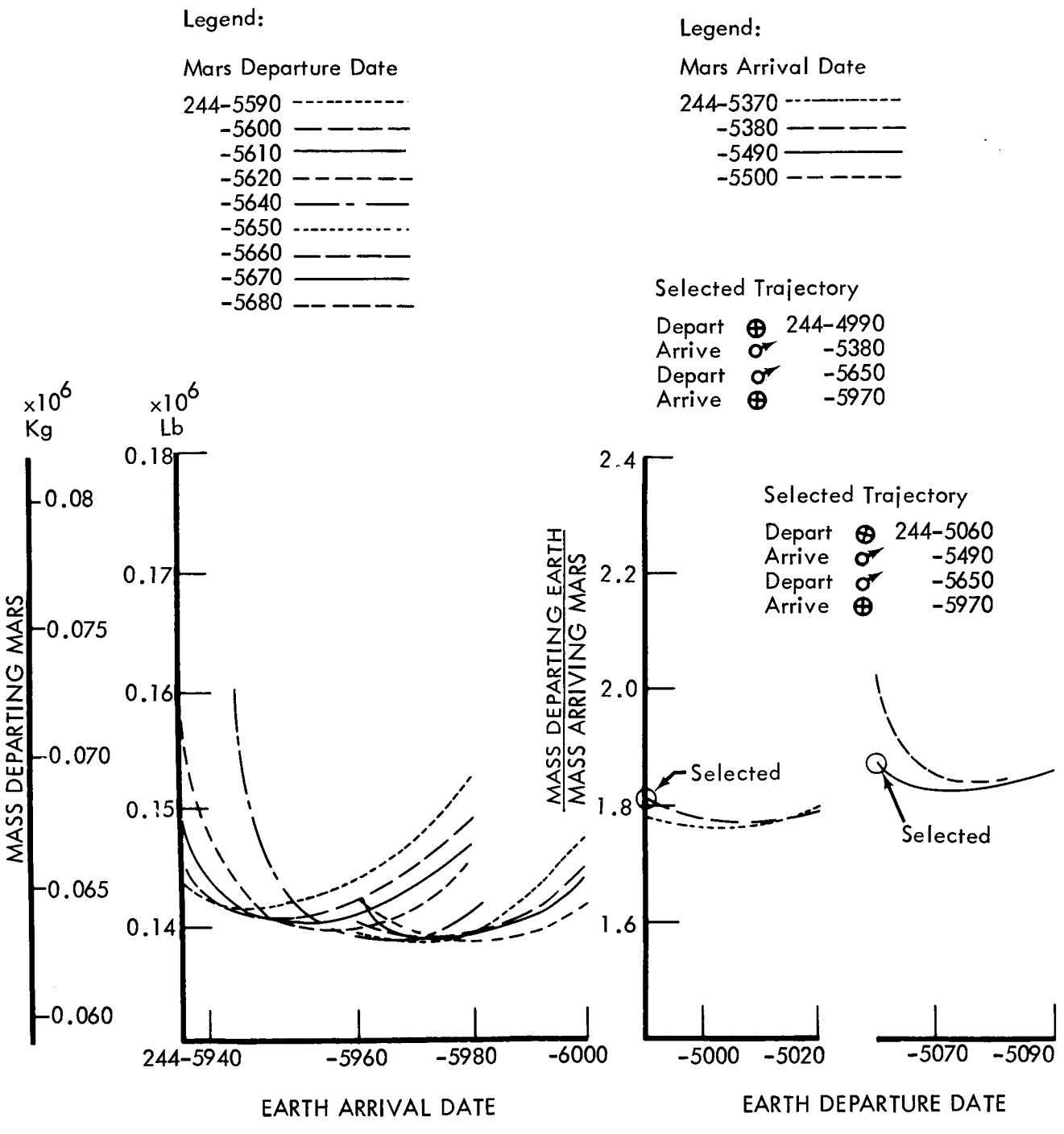


Figure 2.1-21: 1982 MARS LONG OUTBOUND TRIP TRAJECTORIES

Legend:

Mars Departure Date

244-7290 ----------
-7300 — — —
-7310 ————
-7320 — — — —

Legend:

Mars Arrival Date

244-7040 ----------
-7050 — — —
-7220 ————
-7230 — — — —

Selected Trajectory		Selected Trajectory	
Depart ⊕	244-6620	Depart ⊕	244-6670
Arrive ♂	-7040	Arrive ♂	-7220
Depart ♂	-7320	Depart ♂	-7320
Arrive ⊕	-7570	Arrive ⊕	-7570

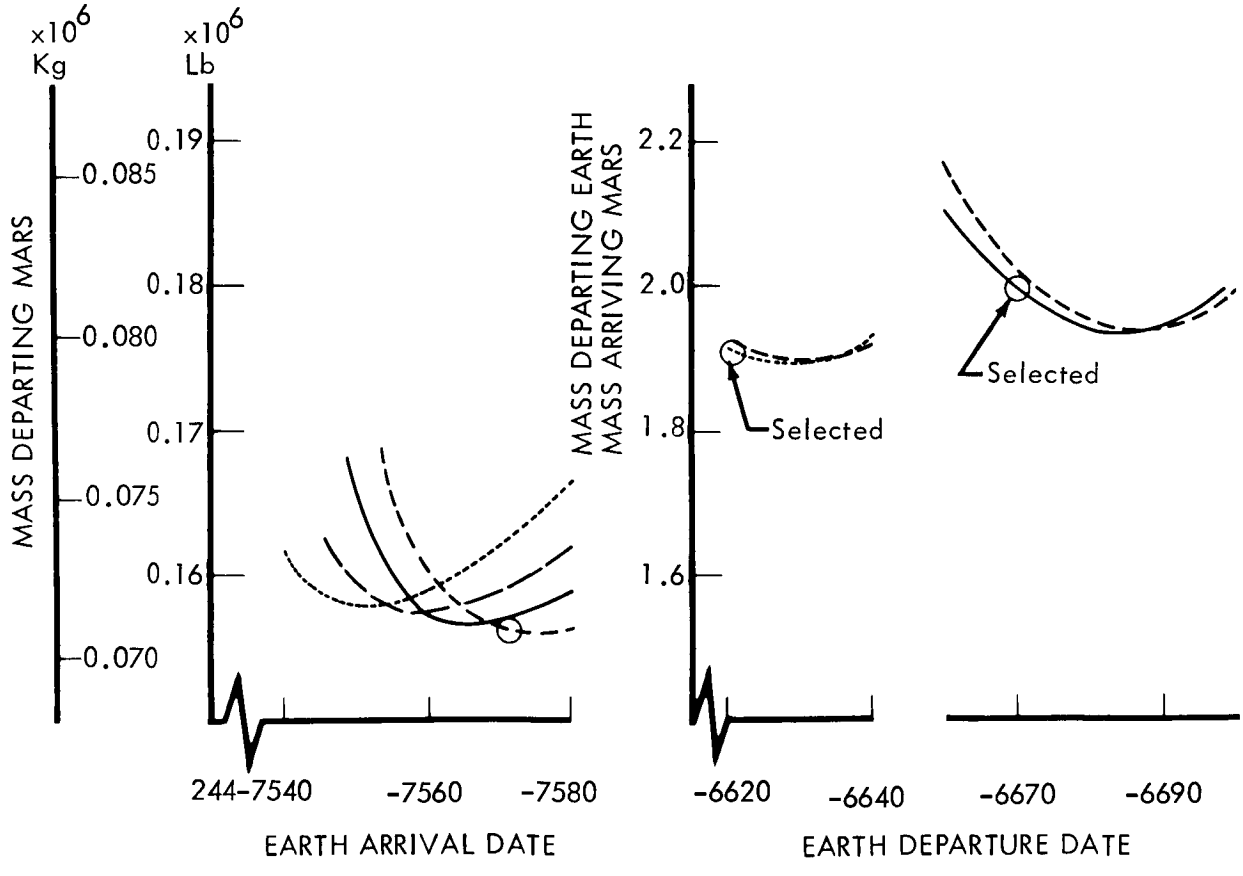


Figure 2.1-22: 1986 MARS LONG OUTBOUND TRIP TRAJECTORIES

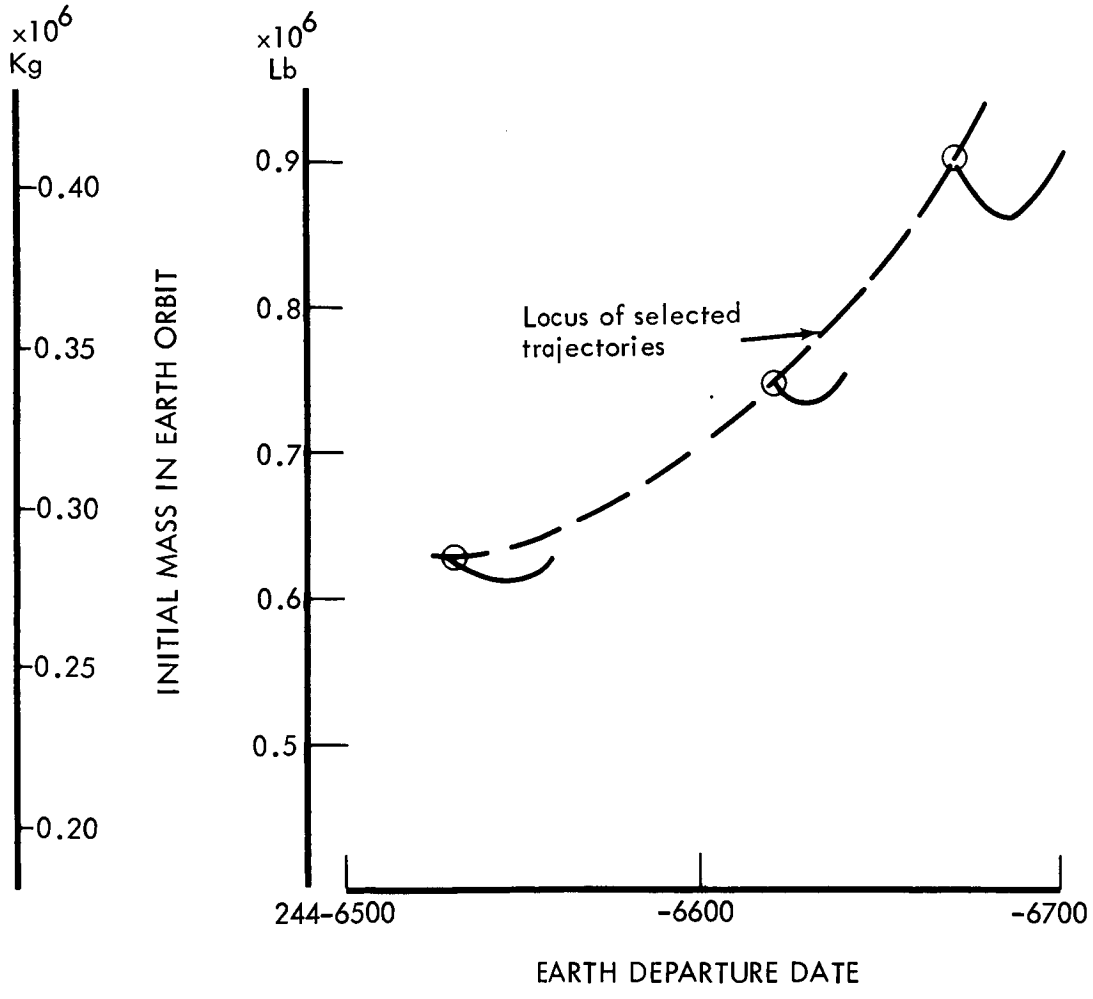


Figure 2.1-23: EFFECT OF EXTENDING LAUNCH BY USE OF LONG OUTBOUND TRIPS — 1986 Mars Trajectories

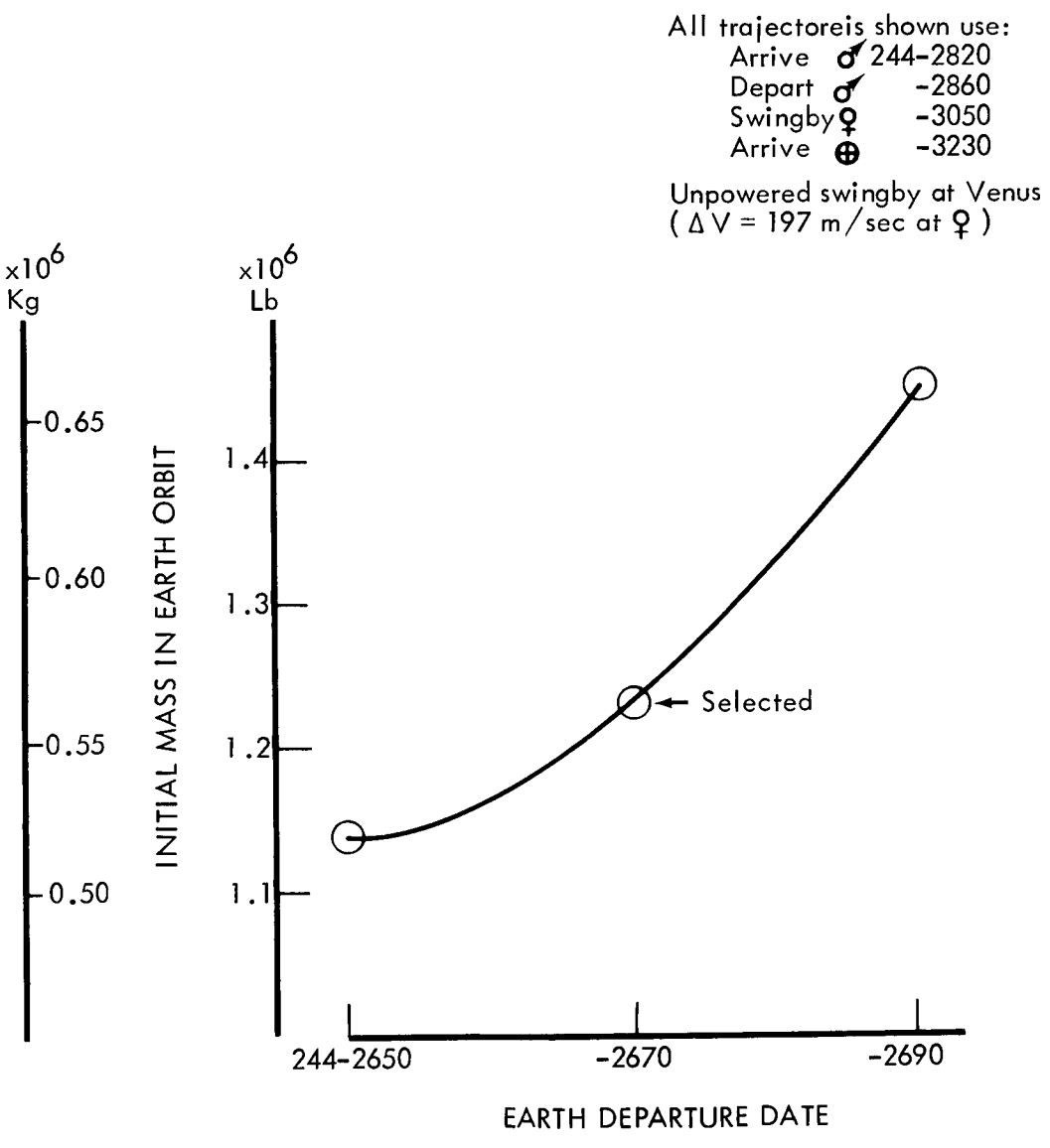


Figure 2.1-29: 1975 MARS TRAJECTORIES WITH VENUS SWINGBY

For each Earth departure date near the minimum IMIEO point, the computer printout was also scanned to find the trajectories (if any) that allowed large ΔV at Venus and required less IMIEO than an unpowered flyby at that Earth departure date. No reduction in minimum IMIEO could be found, although the selection of powered flybys tends to flatten the curve around the minimum. Figure 2.1-29 is typical of the graphs used to select Venus swingby trajectories. Appendix A-3 contains similar data for the other swingby missions.

2.1.3.5 Sensitivity of Trajectory Selections

The nominal trajectories for IMISCD were selected on the basis of initial mass in Earth orbit. To perform the selection required, an initial estimate of the propulsion system specific impulse (I_{sp}), the propulsion stage mass fraction (λ'), and the vehicle module weights was made. A short study was performed to test the sensitivity to mass fraction and to Mars excursion module (MEM) weight of the approach used for the trajectories selection.

For the 1986 Mars opposition class mission, a set of IMIEO calculations was performed for all λ's set equal to 1, and for all $\lambda' = 0.9$ of the original estimates. (See Figure 2.1-30 for a comparison of these, with the results using the original estimated λ'. Figure 2.1-31 shows that in each case, using the standard selection method, the same trajectory was selected as the best nominal for design.) Similarly, for a 1982 Mars opposition class mission (Figures 2.1-32 and 2.1-33), no significant difference in MIMIEO mission selection resulted from the variation of λ'.

A set of IMIEO calculations was performed for the 1986 opposition class mission, with the MEM weight equal to 100,000 pounds rather than 55,000 pounds. Figure 2.1-34 shows that this did not affect the dates selected for a nominal mission.

2.2 SUPPORTING STUDIES

A basic objective of the study was to design a system that would have the flexibility to accommodate a wide variety of missions. This means one that can accommodate the variety of ΔV requirements imposed on the individual ΔV phases of flight by variations in mission date (opportunity) and mission type (i.e., opposition class, conjunction, swingby, etc.). Questions naturally arise about the effect of various alternatives and refinements. These generally fall in two categories: first, alternatives that reduce mission energy requirements, including multiple vehicle modes, alternative interplanetary trajectories, such as perihelion kick and lob trajectories, and elliptical parking orbits at Mars and Venus; and second, refinements and optimizations to account for real-world requirements such as launch window and abort. These latter problems, of course, also directly influence the energy requirements and ΔV distributions. Similarly, when comparing alternative propulsion and aerobraker mixes, a question arises about the degree of optimization employed for each combination. Certain of the alternatives and refinements were investigated to provide additional insight and perspective. These were:

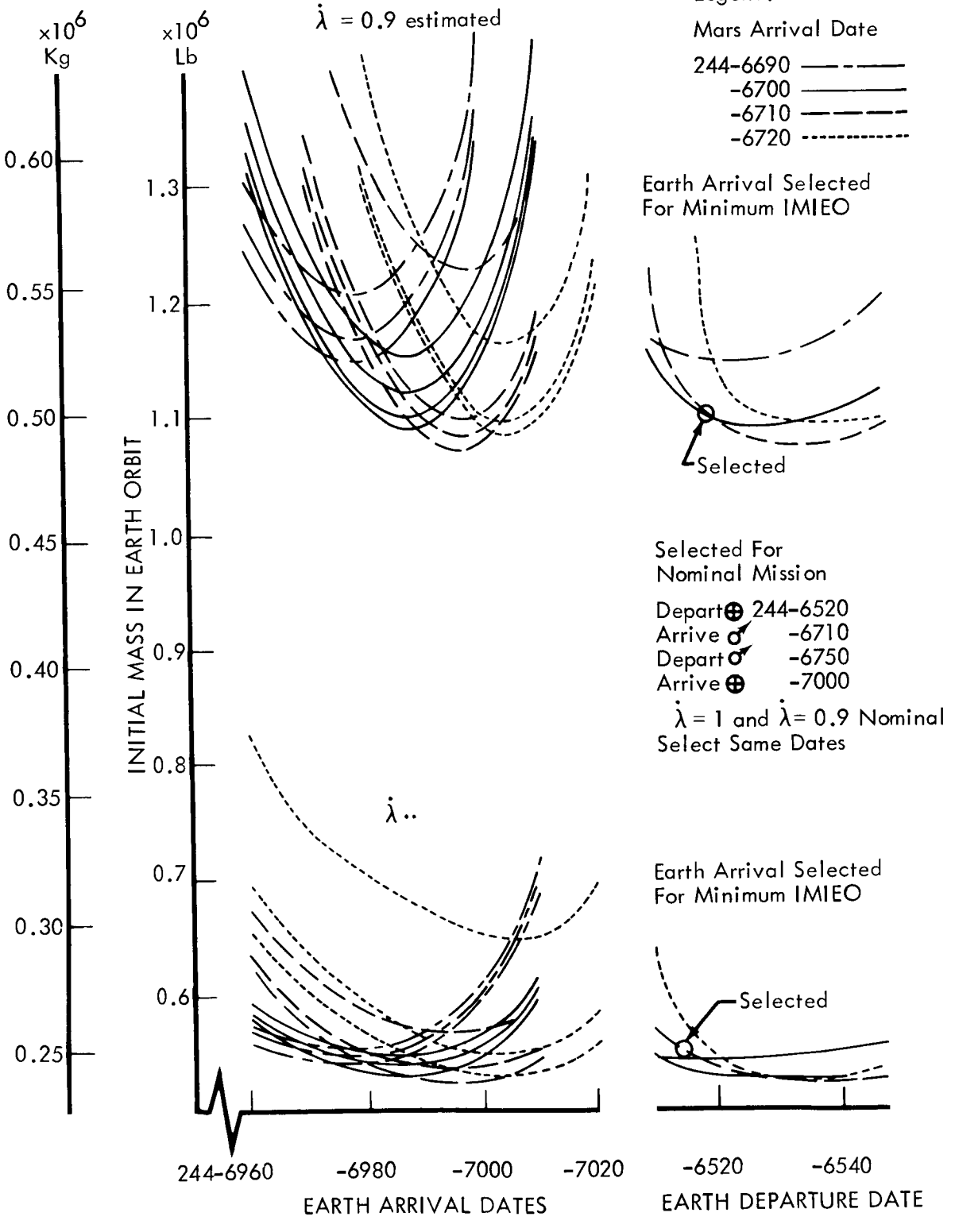


Figure 2.1-30: EFFECT OF EXTREME VALUES OF $\dot{\lambda}$ ON 1986 MARS MISSION

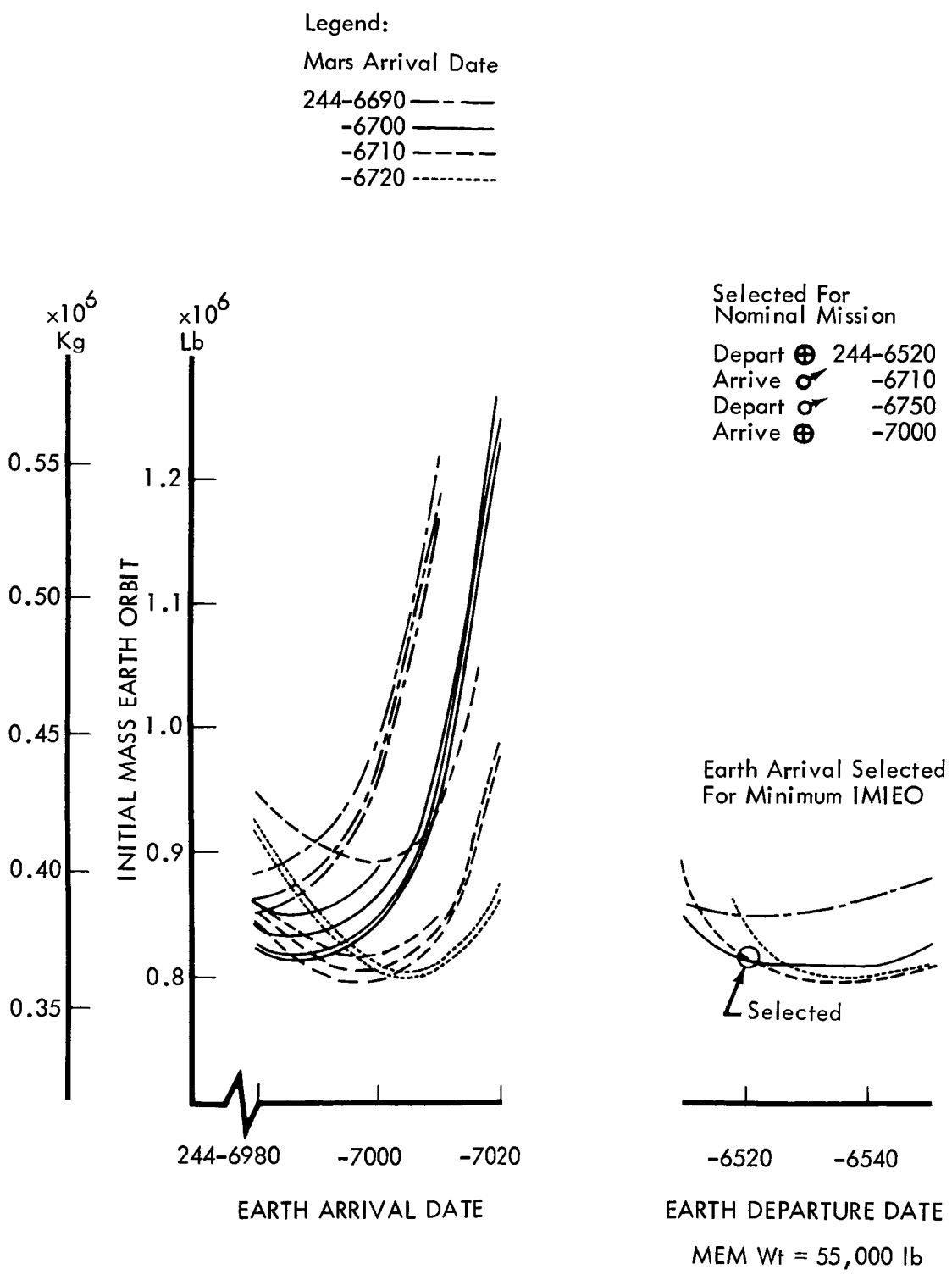


Figure 2.1-31: NOMINAL 1986 MARS OPPOSITION TRAJECTORIES

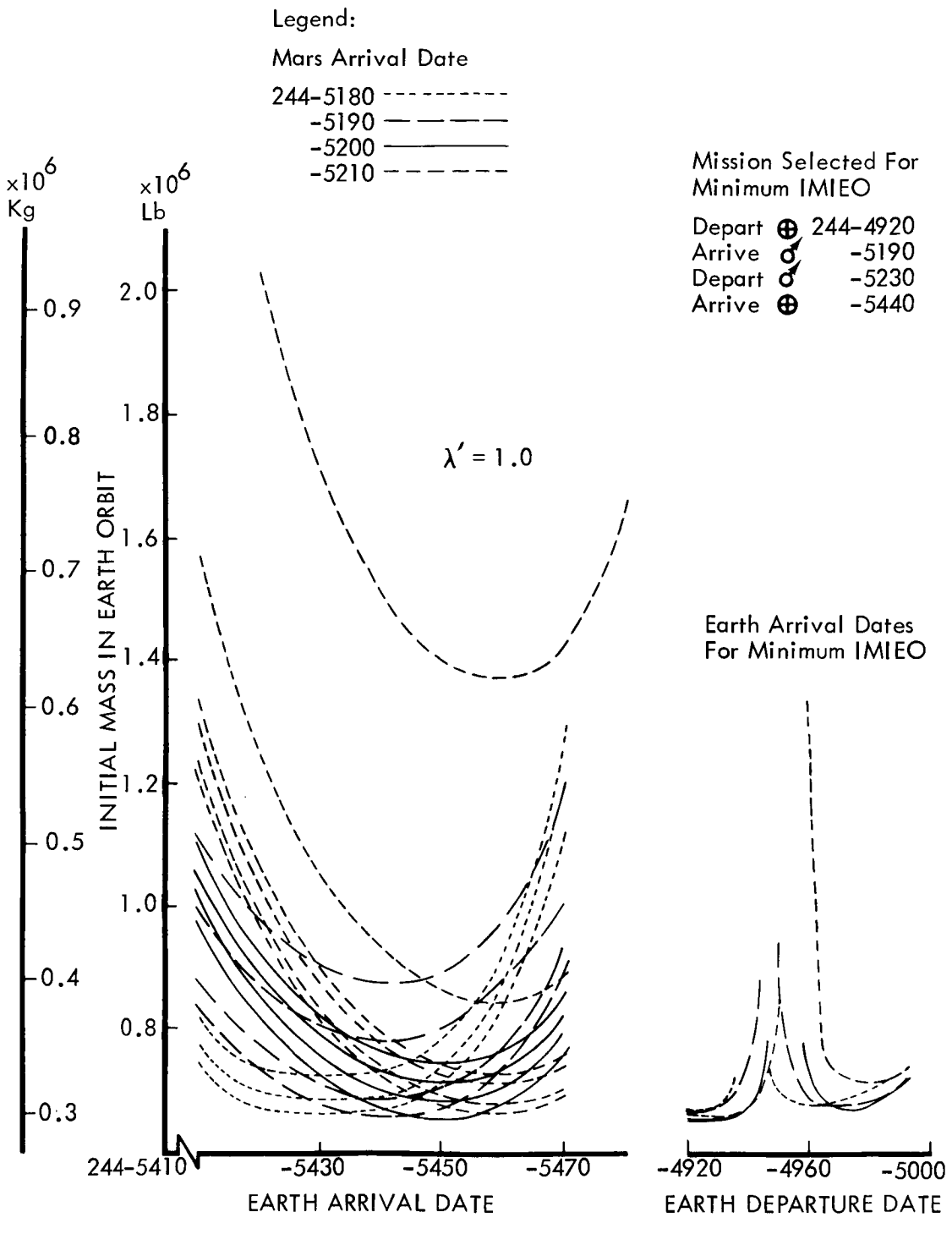


Figure 2.1-32: EFFECT OF HIGH λ' ON 1982 MARS OPPOSITION MISSION

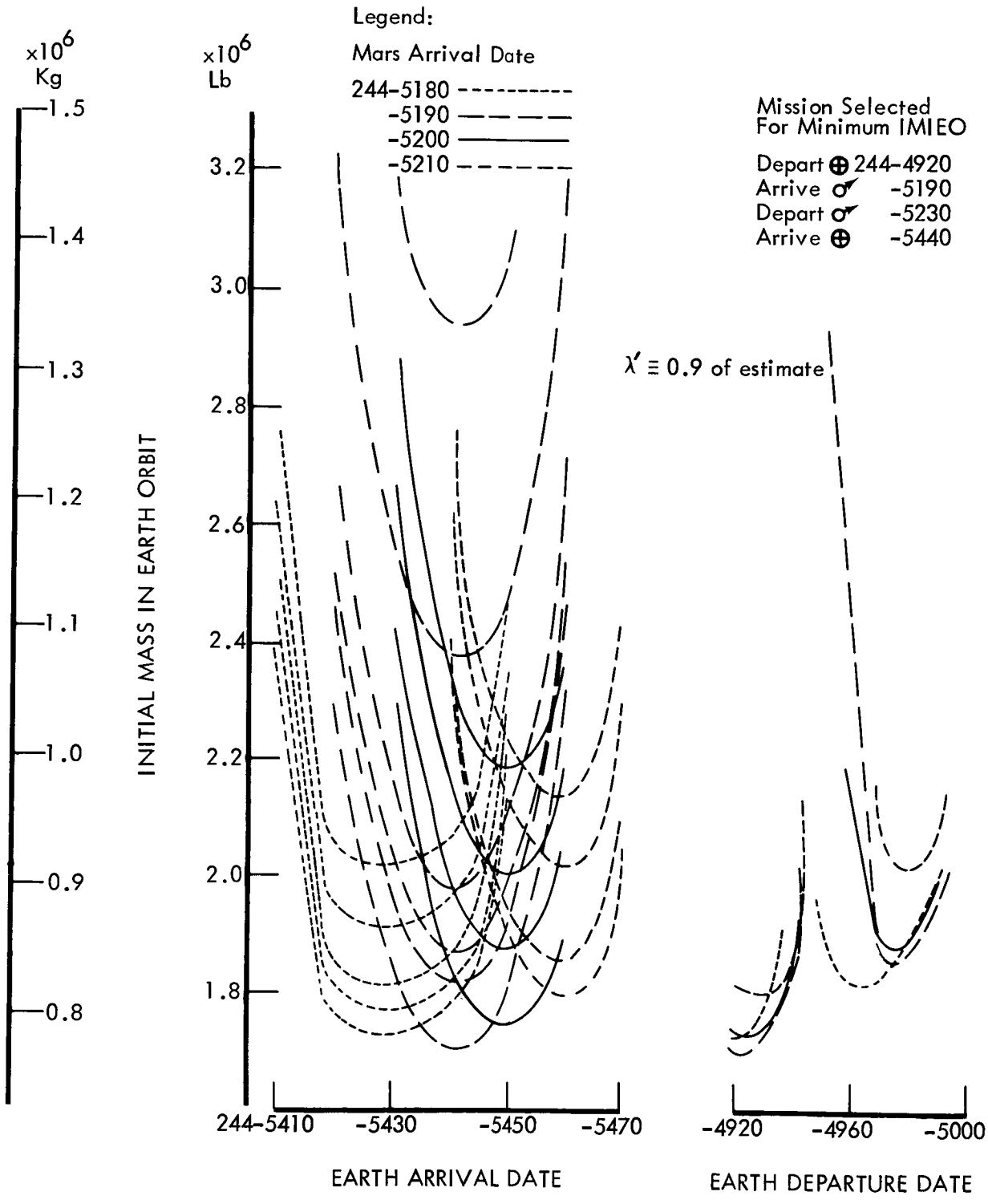


Figure 2.1-33: EFFECT OF LOW λ′ ON 1982 MARS OPPOSITION MISSION

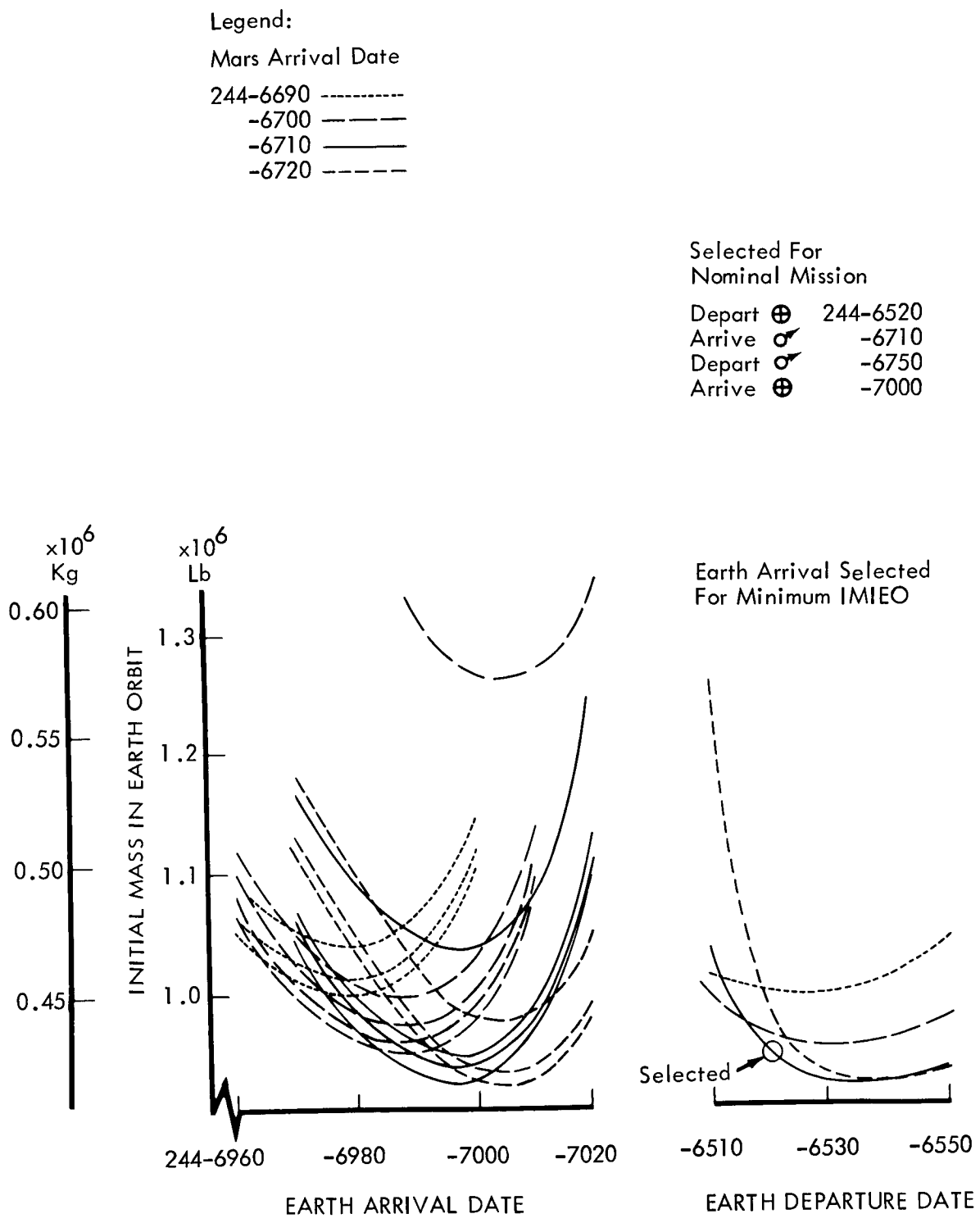


Figure 2.1-34: EFFECT OF HEAVIER MEM WEIGHT (100,000 lb) ON 1986 MARS MISSION

1) Elliptical parking orbits at the planets.

2) Earth departure launch window penalties.

3) Aerobraker trajectory optimization.

These investigations, presented in the following three sections, indicate a future requirement for more detailed analysis of launch window requirements at Earth and the planets, and for clarification of the relationships between planet orbit selection on an energy basis and the scientific mission requirements.

2.2.1 ELLIPTICAL ORBITS ABOUT THE PLANETS

This investigation evaluated the ΔV savings at planet capture and departure through the use of an elliptical orbit about the planet. Interplanetary round-trip capture missions with high velocity requirements can be made more economical through the use of an elliptical parking orbit at the target planet. Planetary capture and departure from a point near periapsis of the elliptical parking orbit can result in significant savings in ΔV compared to the circular planetary orbit. Savings in ΔV are increased as a function of orbit eccentricity. However, when vehicles descend to the surface and return, there is a penalty and an optimization is involved.

The immediate result of the ΔV savings is a reduction in planetary capture and departure stage size with an even greater effect on IMIEO. However, performance improvement can also be measured in terms of a useful payload increase if the stages are to remain sized for the circular orbit requirements. Alternatively, a stage designed to capture in a circular orbit for a particular mission may have insufficient ΔV to circularize if used on a different mission. Capture in an elliptical orbit, however, may reduce the ΔV requirements to the point where the stage can be used. This flexibility could allow for more stage commonality in an interplanetary space program. These potential advantages must, of course, be traded against the scientific requirements at the planet. Despite the substantial saving in IMIEO shown below, the eccentric orbit was treated as an alternative offering additional payload or flexibility for the selected vehicle.

2.2.1.1 Analysis

A primary requirement of round-trip missions is the proper orientation of the asymptotes of the capture and departure hyperbolas (bend angle). This is accomplished in a circular planetary orbit, Figure 2.2-1a, by applying the departure ΔV at a predetermined point on the orbit. The velocity required to depart is the same regardless of the point at which departure is initiated. The greatest ΔV savings with an elliptical planetary orbit is realized when capture and departure occur at periapsis of the eccentric orbit. Proper asymptote orientation, however, is only

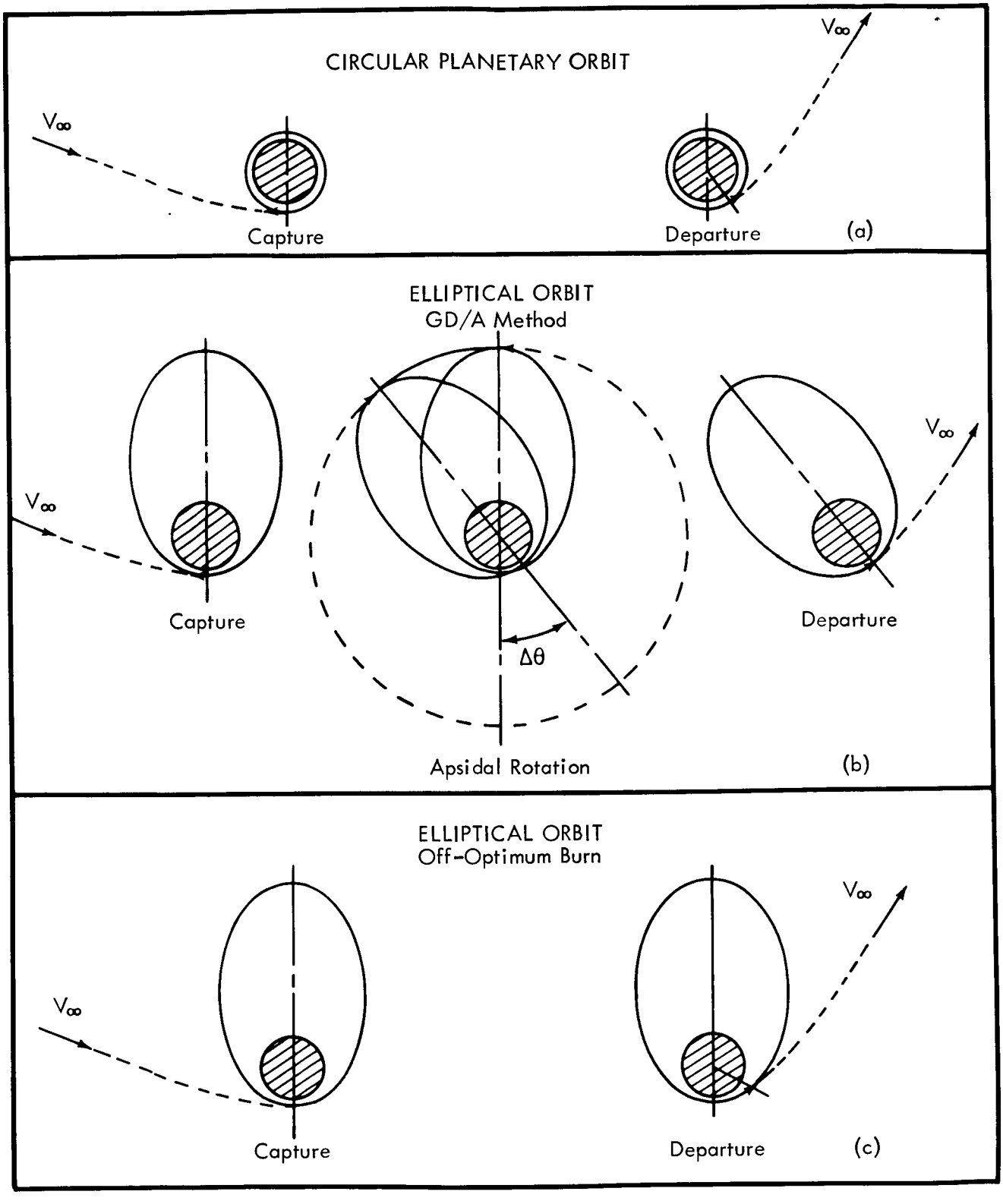


Figure 2.2-1: PLANETARY ORBIT CAPTURE AND DEPARTURE

assured if the orbital line of apsides is rotated prior to departure. A method proposed by General Dynamics/Astronautics* is shown in Figure 2.2-1b and provides for apsidal rotation by using two additional propulsive ΔV's prior to departure. The first impulse at apoapsis circularizes the orbit and a second (identical) impulse decircularizes to the original eccentricity after coasting through a predetermined central angle.

Asymptote orientation and reduced capture and departure ΔV can also be provided by allowing propulsive impulses to occur at a point other than periapsis as in Figure 2.2-1c. This may require that both the capture and departure impulses occur at an off-optimum point in order to achieve the maximum ΔV reduction and the required bend angle between the incoming and outgoing asymptotes. This procedure results in a net ΔV reduction that is superior to the GD/A approach for missions where the ΔV is applied near periapsis. Missions requiring propulsive impulses at true anomalies of about ± 90 degrees may be near the "break-even" point for further performance improvement, although this is only an estimate based on analysis of one set of conditions. Trajectory parameters and propulsion characteristics may alter these results.

Two missions similar to the IMISCD missions were analyzed. The evaluation results are discussed below for a 1986 Mars opposition mission and a 1980 Venus short-stay mission.

Mars 1986 Opposition---An analysis of the ΔV savings available by using an elliptical orbit rather than a circular orbit was performed for the Mars 1986 opposition. This mission was analyzed using trajectory simulations representing a nuclear propulsion system with an initial thrust-to-weight ratio of 0.3 Earth g and a specific impulse of 828 seconds. The Earth-Mars trajectories are based on the following:

Earth Launch:	April 9, 1986 (J.D. 244-6530)
Outbound Leg	180 days
Stay Time	30 days
Mission Duration	460 days

Orbits about Mars are circular at an altitude of 1000 kilometers and elliptical with a periapsis altitude of 1000 kilometers and an eccentricity of 0.7.

*K. Ehricke, GD/A-AOK 63-019, *A Study of Early Manned Planetary-Interplanetary Round Trip Missions*, NASA Contract CR-5548, General Dynamics/Astronautics, July 1963.

Figures 2.2-2a and 2.2-2b illustrate the performance improvement available with an elliptical orbit for this particular mission. Capture and departure ΔV savings of about 975 and 1000 m/sec, respectively, are possible for an eccentricity of 0.7. The total savings available using the GD/A method with an impulsive analysis is only about 700 m/sec for the same eccentricity.

Capture and departure burns occur at or near periapsis for this mission as seen in Figure 2.2-2b because of the bend angle required between the incoming and outgoing asymptotes. These sketches show the planetary orbit plane as parallel to the plane of the paper. Both the incoming and outgoing hyperbolic excess velocity vectors are in the orbit plan. The declination and right ascension of the two asymptotes requires that the orbit plane be inclined 63.1 degrees to the Mars equator with a bend angle between asymptotes of 27.6 degrees. It can be seen in Figure 2.2-2b that a larger bend angle would tend to move the capture and departure burns further from periapsis, thus reducing the amount of ΔV saved.

Venus 1980 Short Stay---An estimate was also made of the performance improvement with an elliptical orbit for the Venus 1980 mission with short stay time. The Earth-Venus trajectory data were based on the following mission times:

Earth Launch	J.D. 244-4310
Outbound Leg	130 days
Stay Time	40 days
Mission Duration	460 days

The elliptical orbit about Venus has a periapsis altitude of 1000 kilometers with an eccentricity of 0.7. The analysis of the Venus mission is based on impulsive velocity increments.

Capture and departure ΔV savings of about 1950 and 1980 m/sec, respectively, are estimated as the performance improvement available for an eccentricity of 0.7. This saving in ΔV is referenced to the capture and departure ΔV requirements for the circular parking orbit, i.e., 4550 and 4060 m/sec, respectively. Using the GD/A method in this application, a total ΔV savings of only 1220 to 1520 m/sec would be realized. The bend angle required between the incoming and outgoing asymptotes is 48.9 degrees. Analysis indicates that the capture and departure burns must be at a true anomaly of about 20 to 30 degrees in order to achieve the required bend angle. This is near enough to periapsis that the savings in ΔV is almost at a maximum, although a total penalty of about 152 m/sec does exist and is reflected in the estimates of the ΔV savings for the Venus mission.

2.2.2 EARTH DEPARTURE PENALTIES

The objective of this investigation was to provide some visibility regarding Earth departure penalties due to orbit precession and variations in the declination of the departure asymptote over a desired orbital launch window.

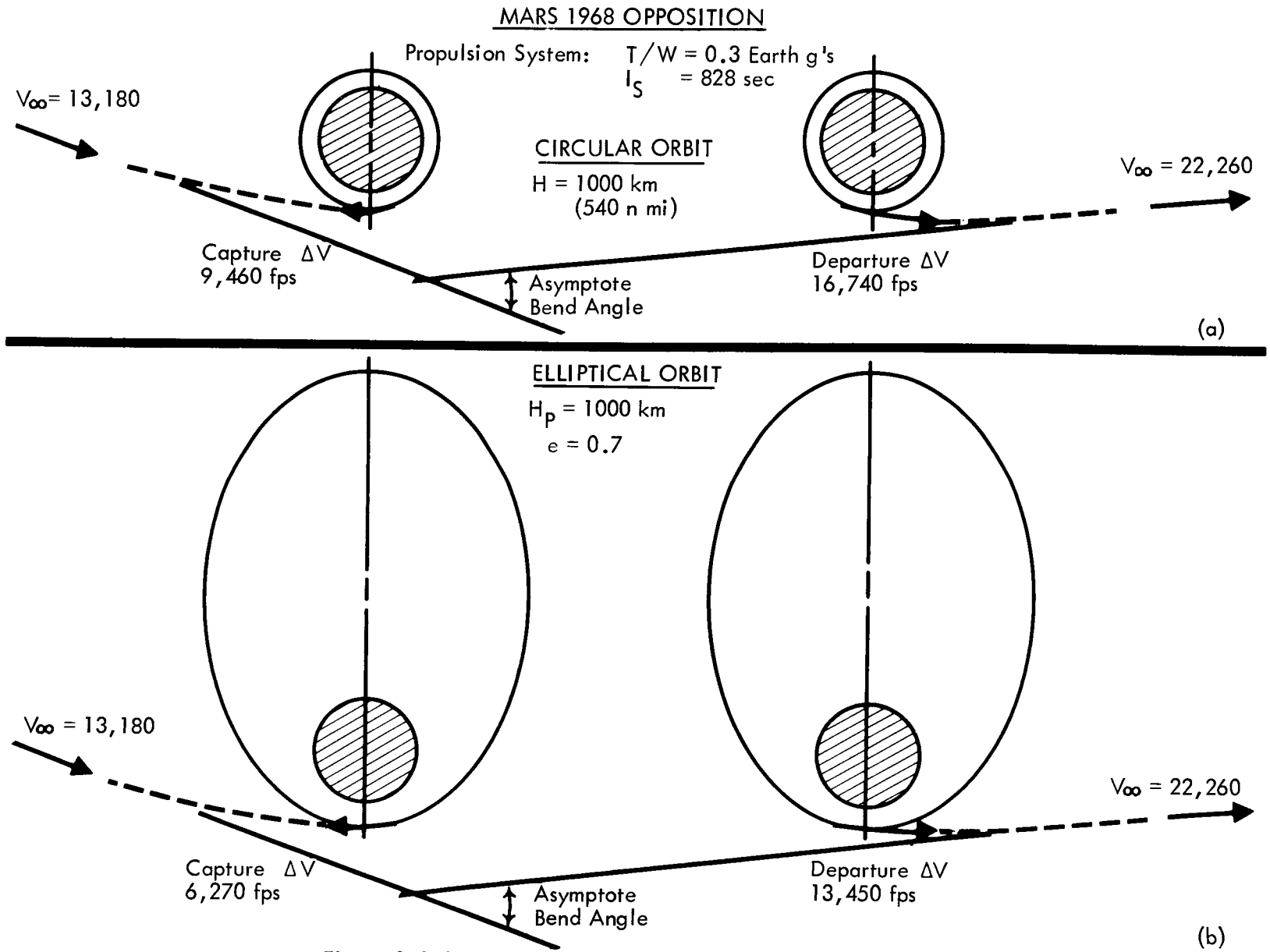


Figure 2.2-2: MARS 1986 OPPOSITION ΔV REQUIREMENTS

Case 2 $|\delta_a| = i$

The expression for sin φ in Case 1 applies but with the second term (following the plus-minus sign) omitted.

Case 3 $|\delta_a| > i$

$$\sin\varphi = \cos i \sin\delta + \sin i \cos\delta \sin(\alpha_a - \alpha - \dot{\Omega} t - n\frac{\pi}{2})$$

$$n = 1,3$$

These expressions are predicated on the assumption that at the nominal departure time, t = 0, the node of the orbit Ω_a is selected to make sin φ zero in Cases 1 and 2 or a minimum in Case 3. The nomenclature is the same as that of Deerwester, et al*, except for i and $\dot{\Omega}$

φ the angle between the outgoing asymptote and its projection on the parking orbit plane. A positive value of sin φ is interpreted thus: The outgoing asymptote vector does not lie between the acute angle formed between the parking orbit plane and the equator. A negative value of sin φ indicates that the outgoing asymptote does lie between those two planes.

i parking orbit inclination, $i \leq \pi/2$

δ declination of the outgoing asymptote

α right ascension of the outgoing asymptote

$\dot{\Omega}$ first order secular regression rate of parking orbit node due to Earth oblateness (J_{20})

Ω orbit node

$( )_a$ the quantity ( ) associated with the nominal departure time, t - 0

t time

The two solutions called for by the plus-minus term in the expression for sin φ of Case 1 suggest that there are two possible parking orbit nodes at t = 0 that result in φ = 0. They are given by:

$$\Omega_a = \alpha_a - \sin^{-1} \text{cotitan } \delta_a$$

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*Deerwester, J.M., J.F. McLaughlin and J.F. Wolfe, "Earth Departure Plane Change and Launch Window Considerations for Interplanetary Missions," *Journal of Spacecraft and Rockets*, February 1966

A velocity penalty is incurred whenever the Earth parking orbit plane does not contain the outgoing hyperbolic asymptote. When the declination of the asymptote is less (in absolute value) than the inclination of the Earth assembly orbit, the initial liftoff time and azimuth can be selected so that the resulting parking orbit contains the outgoing hyperbolic asymptote at the nominal departure date. If the declination at the nominal departure date is greater (in absolute value) than the available inclination from Cape Kennedy, then the angle between the parking orbit plane and the asymptote may be minimized, minimizing the ΔV penalty. The major difficulty in the problem stems from the fact that the parking orbit precesses rapidly due to the Earth's oblateness, compared to the spatial change in the direction of the required outgoing asymptote. This means that if the departure is delayed by 24 hours past the nominal date, the parking orbit plane will have regressed by about 6 to 7 degrees. Necessarily then, it cannot contain the outgoing asymptote. Therefore, the angle between the asymptote and the orbit plane is larger than the minimum that would have occurred had the departure occurred on time. Such delays can impose either serious ΔV penalties or reduction in effective launch window duration. The sophistication required for the penalty calculations limited the scope of this analysis to one representative mission. For that case, a 1982 Mars opposition mission, the ΔV provided by the arbitrary allocation used in the study provides a launch window of only 16 days, compared to the desired 30-day window.

Although the arbitrary allowance of about 10% IMIEO used in the study may be less than adequate, further iteration was not justified, in view of the complexity of a total optimization of energy management, including consideration of launch window at the planets.

Some additional aspects of launch window requirements are discussed in Section 2.2.3.

2.2.2.1 Analysis

The ΔV penalty due to off-nominal departure times can be computed as a function of departure date. The first step is to compute the required plane change angle φ. Three distinct cases can be differentiated: the required declination (absolute value) of the outgoing hyperbolic asymptote at the nominal departure time is less than, equal to, or greater than the inclination of the Earth parking orbit. The derived expressions are:

Case 1 $\quad |\delta_a| < i$

$$*\sin\varphi = \cos i \left[\sin\delta - \tan\delta_a \cos\delta \cos(\alpha_a - \alpha - \dot{\Omega} t)\right]$$

$$\pm (\sin^2 i - \cos^2 i \tan^2 \delta_a)^{1/2} \cos\delta \sin(\alpha_a - \alpha - \Omega t)$$

*This expression for $\sin\varphi$ differs from that given in Deerwester's work, Eq. (1). The numerical difference between the equations becomes significant when $|i - |\delta_a||$ is not small. In the numerical example that follows, $i - |\delta_a| = 6.6°$ and spot checks indicated no significant numerical difference in $\sin\varphi$ between the two expressions.

The sign rule is to use the plus sign if the selected Ω_a is such that

$$\Omega_a - \frac{\pi}{2} \leq \alpha_a \leq \Omega_a + \frac{\pi}{2}$$

Otherwise use the minus sign.

In Case 3 use n = 1 when $\delta_a > 0$ and n = 3 when $\delta_a < 0$.

A numerical example has been computed for φ and the corresponding ΔV plane-change penalties associated with the following interplanetary mission:

1982 Mars Opposition Mission

Earth departure date (J.D.) 244-4927 (t = 0)

Outbound leg trip time 280 days

Stay time at Mars - 40 days
Total trip time - 530 days } This information is not used.

The following assumptions were made to facilitate the calculations:

1) The required declination as a function of departure date is taken from the computer program printout wherein the outbound leg trip time is held constant at 280 days. More rigorously, the outbound trip time should be permitted to vary in order to optimize some given performance index. The variation of the required declination with departure date is given in Figure 2.2-3. The digital computer output values are shown at intervals of 40 days.

2) The required right ascension is equal to the right ascension at t = 0, $(\alpha_a - \alpha = 0)$. The general trends are not significantly affected by this assumption.

3) The parking orbit plane regresses at the rate

$$\dot{\Omega} = -\frac{3}{2} \frac{n J_{20} \cos i}{a^2 (1 - e^2)^2}$$

Taking a = 1.084 Earth radii

e = 0

$GM_\oplus = 3.986 \times 10^5 \text{ km}^3/\text{sec}^2$

$J_{20} = 0.0010823$

we obtain

n = 5423 deg/day

and

$\dot{\Omega} = -7.48 \cos i$ deg/day

4) The parking orbit inclination

i = 28.5 deg

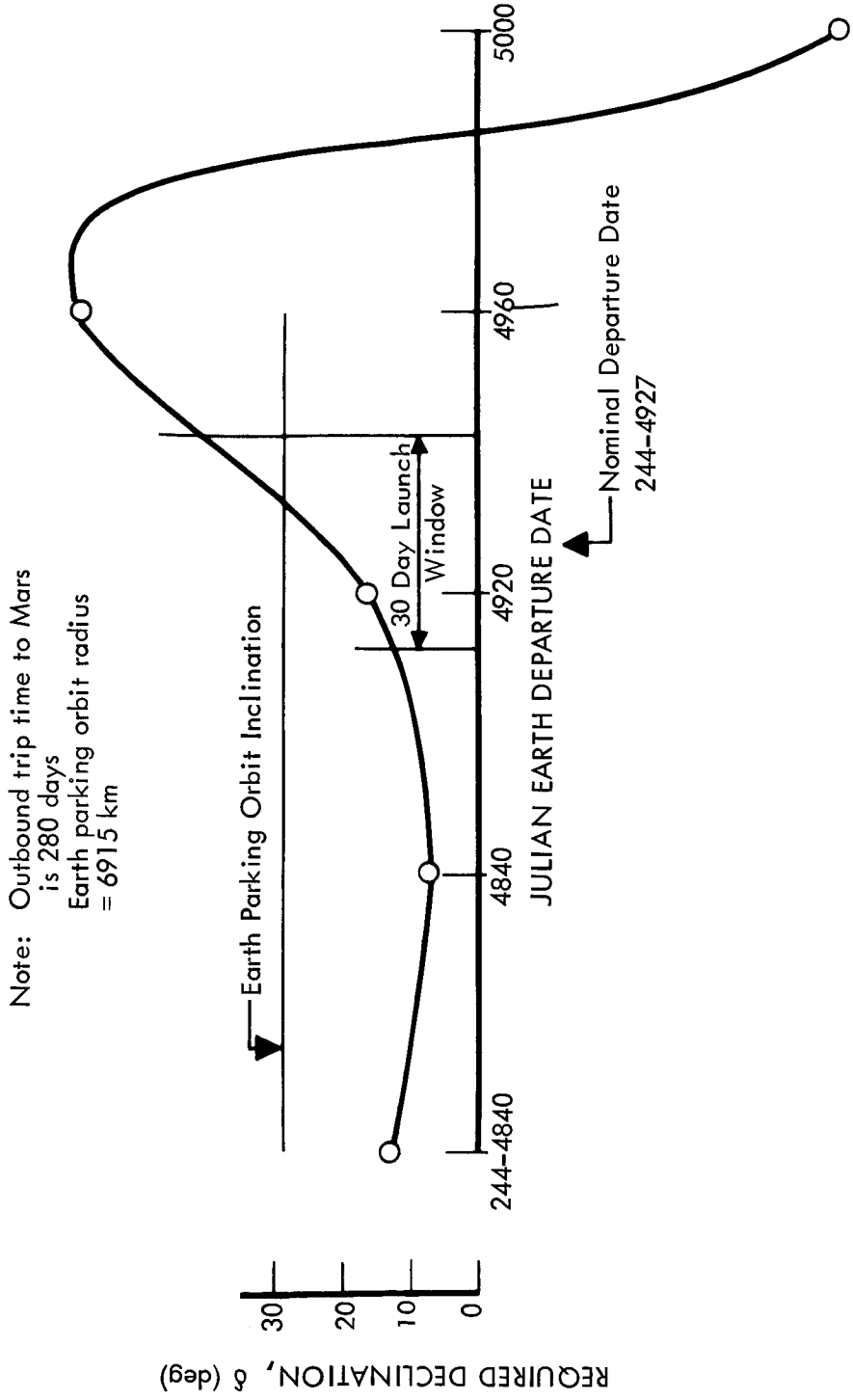


Figure 2.2-3: REQUIRED DECLINATION OF OUTGOING HYPERBOLIC ASYMPTOTE

The variation in the required plane change angle φ as a function of departure date is given in Figure 2.2-4. Two curves are shown corresponding to the two possible values of Ω_a designated by Ω_a^+ and Ω_a^- in the interval 15 days before to 15 days after the nominal departure date. Note that the Ω_a^+ curve does not seem to yield a negative φ. This may be attributed to the particular variation of the required δ in this interval.

The ΔV penalty arising from the required plane change φ may be computed in a variety of ways. Two modes of departure considered in this analysis (Mode 2, dual impulse, and Mode 3, single impulse) were defined and studied previously in the Deerwester study. Mode 2, illustrated in Figure 2.2-5, consists of injecting into an escape hyperbola in the plane of the assembly orbit with a hyperbolic excess velocity V_{HP}, less than the required V_{HP}. At some distance (effectively infinity), the necessary plane change φ is made in addition to acquiring the remainder of the required excess velocity. This mode requires two impulses.

Mode 3, illustrated in Figure 2.2-6, combines the basic energy impulse and plane change impulse into one maneuver. In this mode the necessary plane change and energy velocity input require a vectorial summation of the increments and a proper selection of the ignition point. Single-impulse technique, Mode 3, gives the ΔV penalty curves shown in Figure 2.2-7. These curves, shown solid, correspond to Ω_a^+ and Ω_a^-. The two-impulse technique, Mode 2, results in the penalty ΔV shown by the dashed curves.

The variation of required hyperbolic velocity at departure V_{∞_1} and the corresponding incremental velocity from the given Earth circular orbit ΔV_1 are shown in Figure 2.2-8.

To more clearly visualize the Earth departure requirements, the departure penalties were added to the basic ΔV_1 requirements over a 30-day launch window of Figure 2.2-8 and are shown in Figure 2.2-9. The lower curve shows the simple variation in ΔV_1 over a 30-day period for the 1982 Mars opposition mission with a 280-day outbound leg and a nominal departure date of JD 244-4927. The two upper curves represent the total departure ΔV_1 requirements (i.e., departure penalties plus the nominal injection ΔV_1) for the Mode 2 and Mode 3 types of injection. It can be clearly seen that if a 30-day launch window is required, Mode 2 would imply a 5.11 km/sec ΔV_1 requirement whereas the Mode 3, single-impulse method would necessitate 6.86 km/sec ΔV_1. The initially selected set of trajectories for the 1982 Mars opposition mission established a 3.99 km/sec ΔV_1 requirement. That selection was made on the basis of finding the minimum IMIEO departure and then backing up the IMIEO versus departure date curve by an arbitrary 10% to provide a margin for launch window.

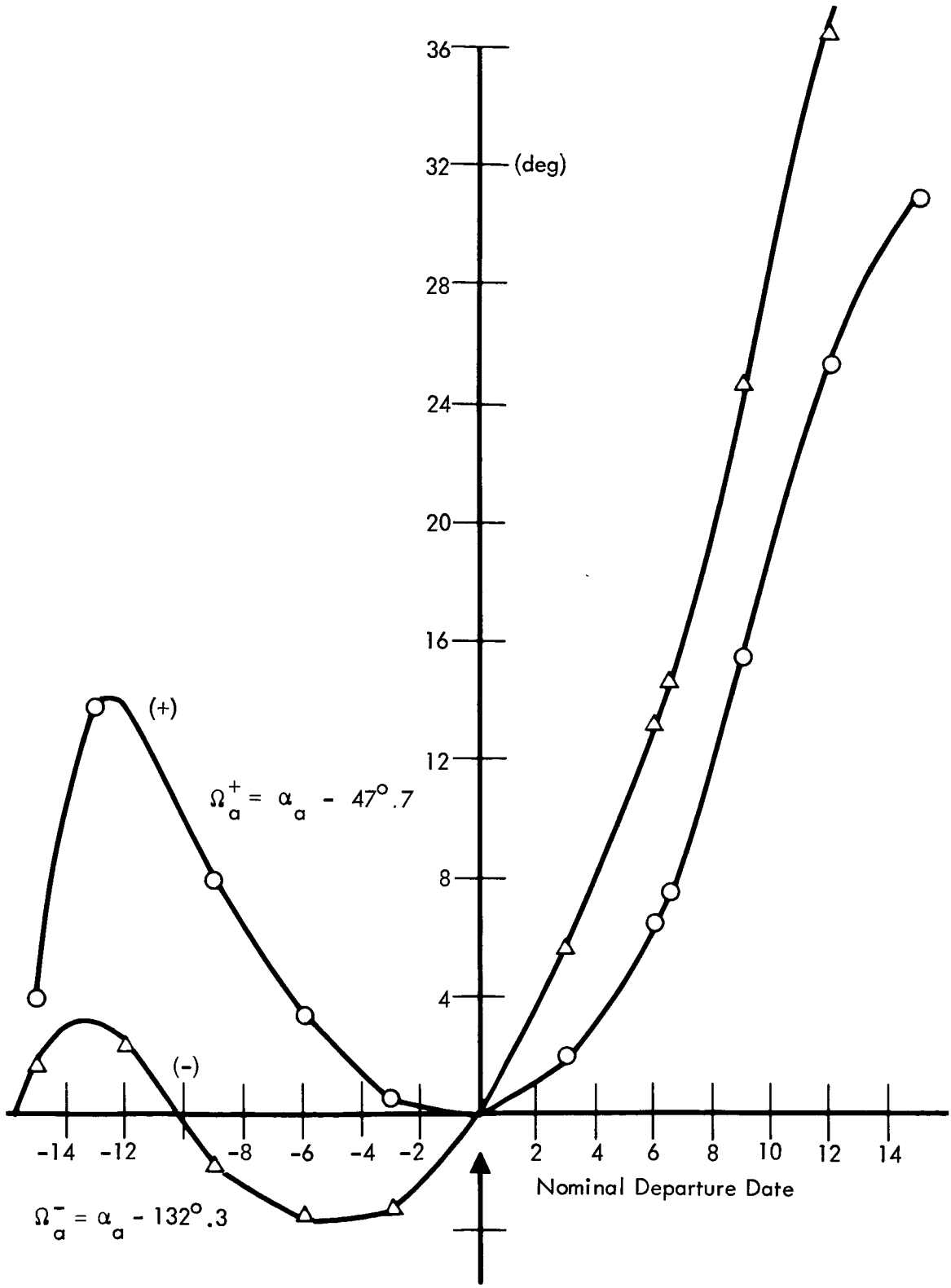


Figure 2.2-4: REQUIRED PLANE CHANGE ANGLE

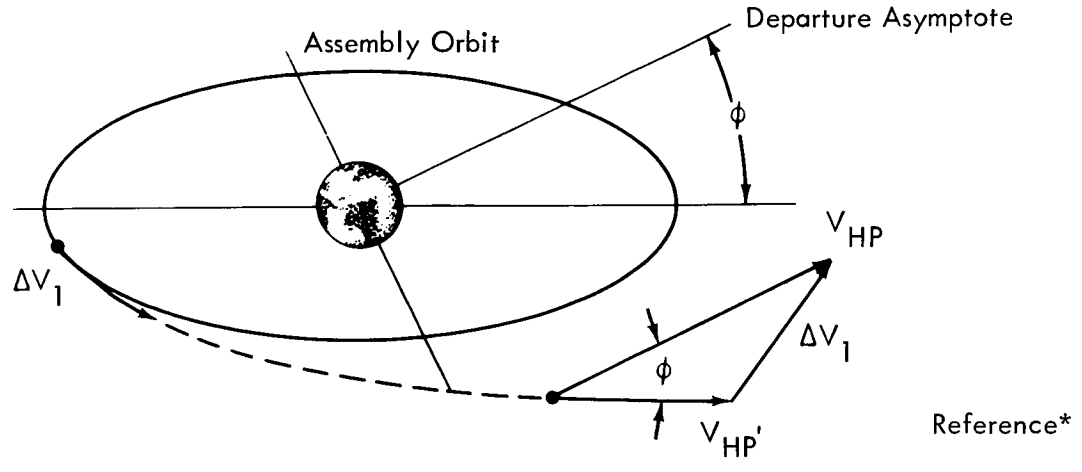


Figure 2.2-5: MODE 2 — Two-Impulse Departure

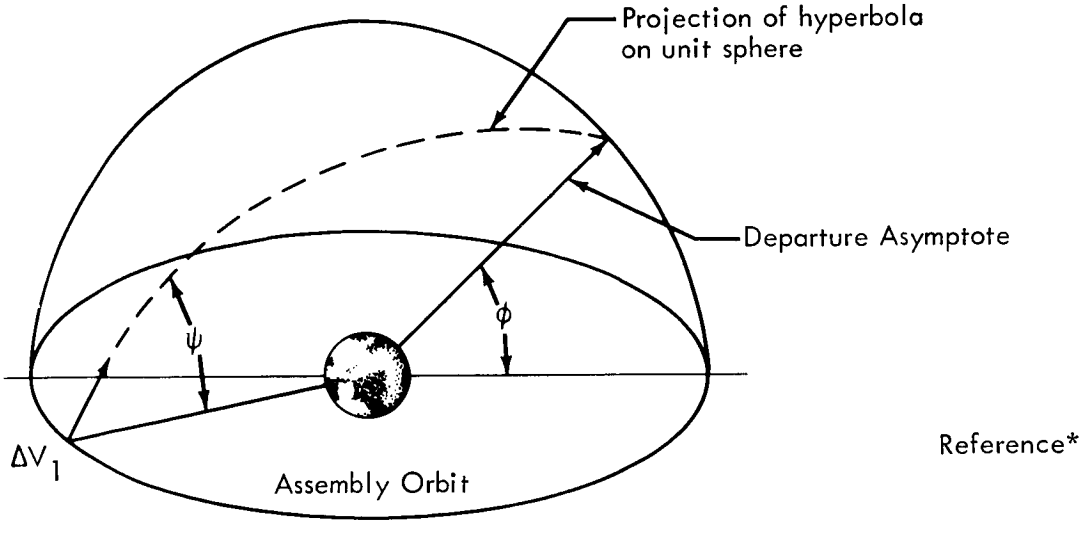


Figure 2.2-6: MODE 3 — Single - Impulse Departure

* Deerwester, J. M., J. F. McLaughlin, and J. F. Wolfe, "Earth Departure Plane Change and Launch Window Considerations for Interplanetary Missions," February 1966.

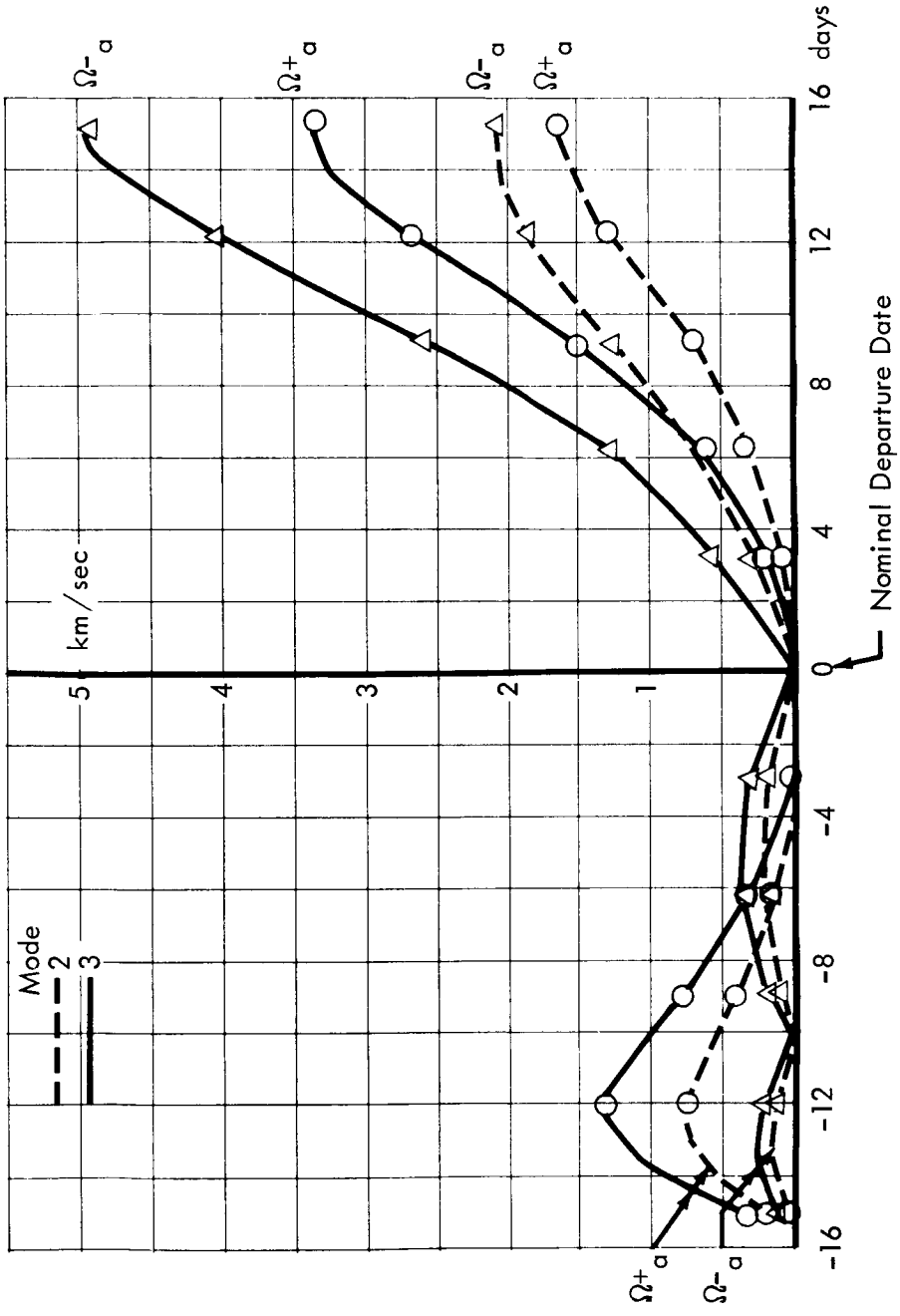


Figure 2.2-7: REQUIRED PLANE CHANGE ΔV

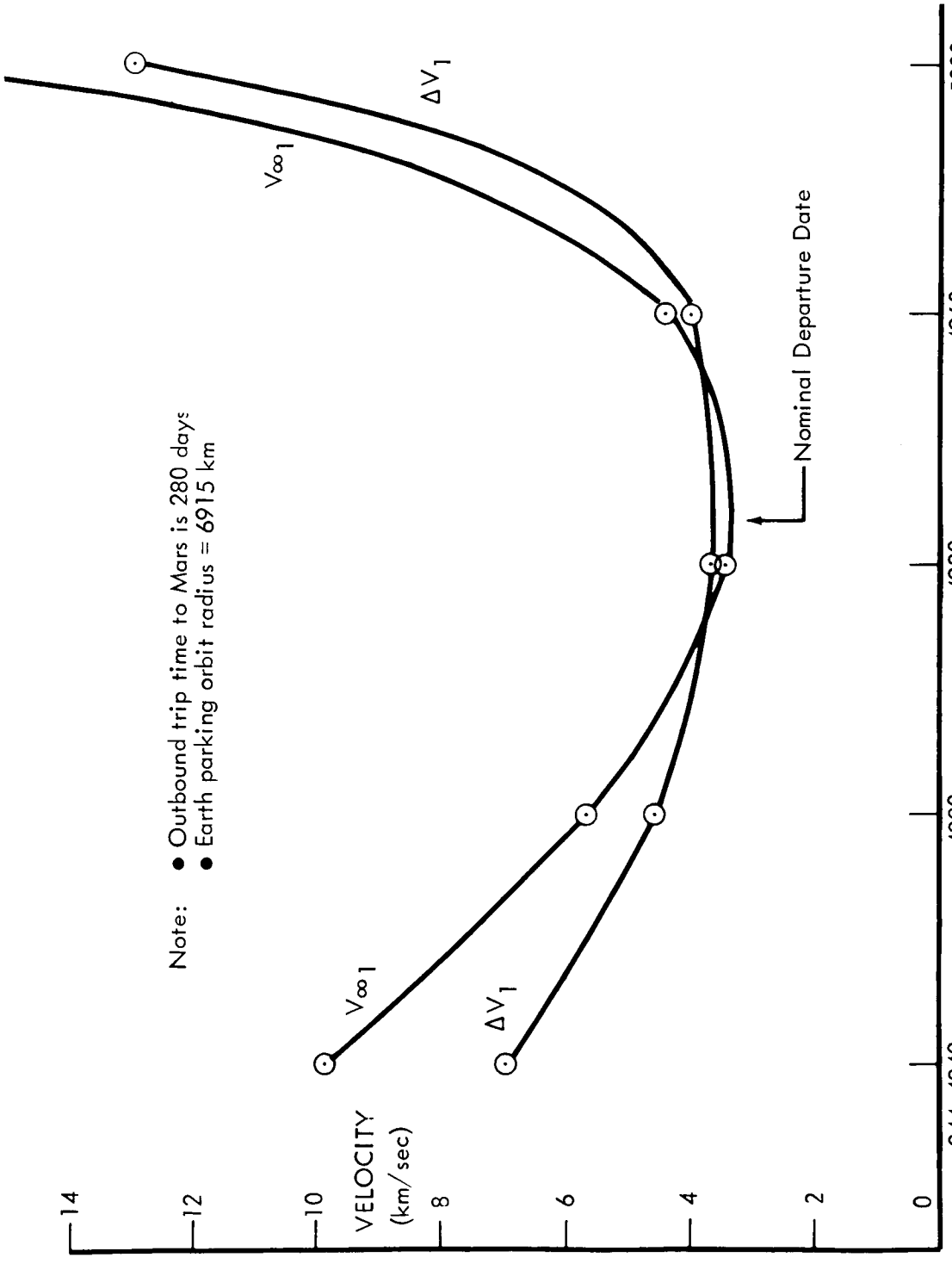


Figure 2.2-8: REQUIRED V_∞ AND ΔV AT EARTH DEPARTURE

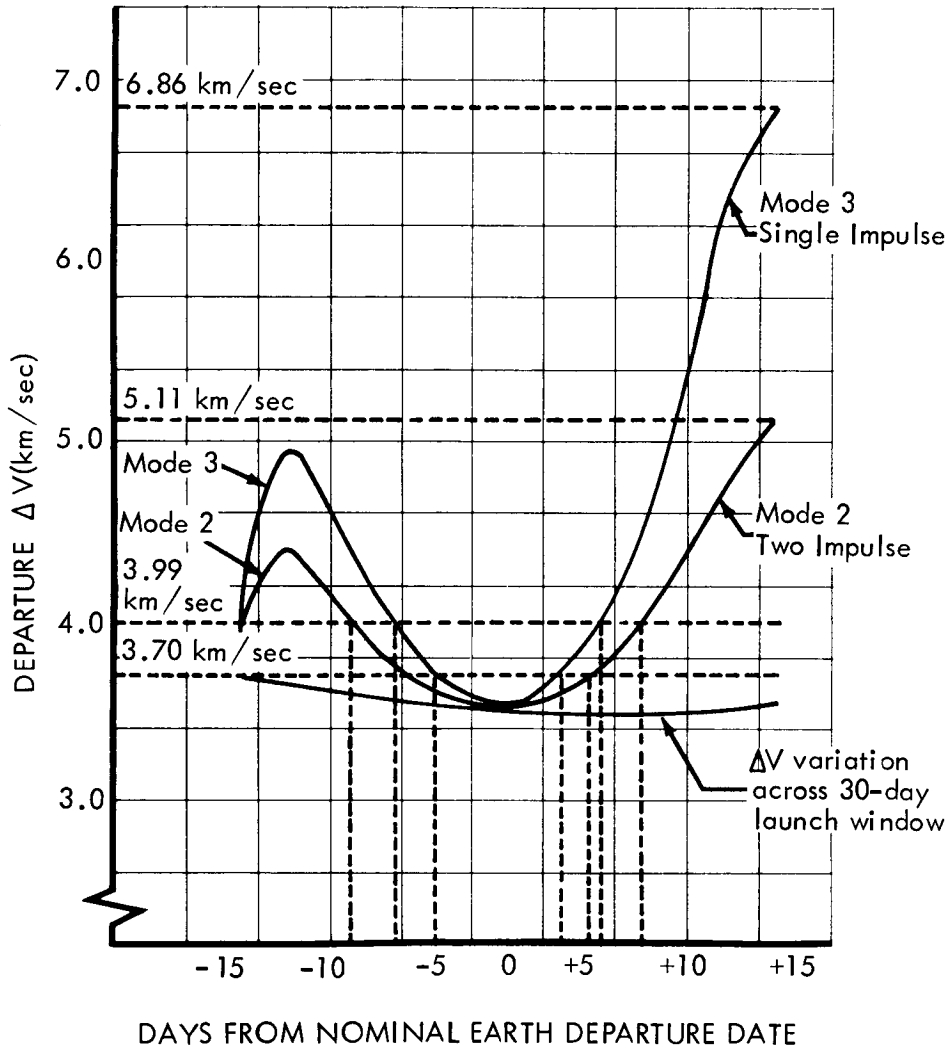


Figure 2.2-9: 1982 MARS OPPOSITION — 530/280 Mission - 40-Day Stay

With that ΔV_1 capability designed into a space vehicle, it can be seen that this would only allow a 12-day launch window for a Mode 3 departure or a 16-day window for Mode 2. If the 3.7 km/sec ΔV_1 were provided, ignoring the plane change penalties and choosing a 30-day launch window, the actual achievable launch windows now become 7 days for the single-impulse Mode 3 and 11 days for the dual-impulse Mode 2. It should be noted, however that an additional discrete opportunity exists at about -15 days from the nominal and that the entire pattern will repeat in about 55 days if there are trajectories with similar ΔV requirements.

A similar geometric situation exists at the target planet. In the case of Mars, which is more oblate than Earth ($J_{200} \simeq 0.00203$), the regression rate would be about 16 cos i deg/day for a close-in orbit. For longer period orbits, the regression would be less, but eccentricity increases the regression rate. If little or no plane change ΔV penalty is to be incurred when departing Mars, the injection point into the Mars orbit must be correctly selected such that the stay or parking orbit will contain the required outgoing asymptote from Mars at departure. If the regression were zero, clearly the parking orbit must contain both the Mars approach and departure hyperbolic asymptotes. With finite regression, the inclination must be such that at the end of the nominal stay time, the parking orbit contains the outgoing asymptote. Since the regression rate is itself a function of the inclination, an iterative solution for inclination is called for. Unfortunately, this penalty consideration and the corresponding allowable departure window duration may compromise the parking orbit inclination and period from the standpoint of scientific worth.

2.2.3 AEROBRAKING MISSION TRAJECTORIES

The selected trajectories of Section 2.1.1 are more or less off optimum for different mixes of chemical and nuclear propulsion and aerobraking at the planets and at Earth. There is, therefore, some degree of approximation involved when the selected trajectories are used to compare various space acceleration systems. Since vehicle combinations using an aerobraker at the planets would see the largest penalty when constrained to fly optimum all-propulsive trajectories, some optimized aerobraker trajectories were derived. Although substantial differences do occur for some missions, the degree of approximation using the trajectories of Section 2.1.1 did not affect the system trade studies. This is illustrated below:

	NAC	NNN
IMIEO Range (10^6 lb)		
used in trade studies:	1.2 to 2.3	1.4 to 1.9
(Section 2.1.1, "Selected Trajectories")		
With reoptimized trajectories	1.1 to 2.2	1.4 to 2.1

See Volume IV for discussion of trade studies.

2.2.3.1 Analysis

Aerobraker trajectories were specifically selected for six missions (i.e., Mars 1982 opposition, Mars 1986 opposition, Mars 1986 conjunction, Mars 1982 with Venus swingby, Venus 1980 with short stay time, and Venus 1980 with long stay time).

1) Sets of trajectories were computed for each of the six missions, varying the trip times and departure dates (and the planet stay times for the conjunction and long-stay missions) over ranges encompassing those values of the parameters established for the initially selected missions (Section 2.1.1).

2) The computer data were reviewed and trajectories of near-minimum total ΔV were identified.

3) Allowable entry velocities for aerobraking of 10 km/sec at Mars and 12 km/sec at Venus, for 100 kilometer altitudes, establish allowable hyperbolic approach velocities of 8.7 km/sec and 6.5 km/sec for Mars and Venus, respectively. Plots of V_{HP_2} were made for the various outbound trip times as a function of departure date to define acceptable trajectories for the aerobrakers. Figure 2.2-10 shows such a plot for the 1982 Mars opposition mission.

4) Total ΔV was then plotted against Earth departure date for each of the apparent near-minimum ΔV trajectories. Figure 2.2-11 shows the 1982 Mars opposition plot for seven possible missions with near-minimum total ΔV's. This plot distinguishes those portions of the curves that would involve excessive V_{HP_2} for the aerobraker missions. From this plot the particular combinations of trip times that offer lowest energy requirements within the allowable limits of V_{HP_2} can be determined. Note that the low ΔV mission here appears to be the 450/200-day mission with a J.D. 244-4940 departure date. A similar plot of Figure 2.2-12 for the all-propulsive case shows a 530/280-day mission to be desirable from a minimum ΔV standpoint with a departure date of J.D. 244-4906.

5) Plots of Figures 2.2-13 through 2.2-17 show variations of incremental ΔV's with Earth departure date for the mission trajectories of interest. From these plots the importance of adjusting the ΔV requirements to accommodate a desired launch window is apparent. These plots, with ΔV_2, are applicable to the all-propulsive missions as well as the aerobrakers.

6) To adjust the ΔV requirements to accommodate a 30-day launch window (strictly from a mission energy standpoint, not considering Earth departure penalties due to orbit precession and change in declinations of the departure asymptote), nominal launch dates were selected across the range of interest, and the highest incremental ΔV's over the $\pm$ 15-day range were selected. Totaling these increments provides a total adjusted ΔV which was then plotted against Earth departure date as shown in Figure 2.2-18 for the 1982 Mars opposition mission.

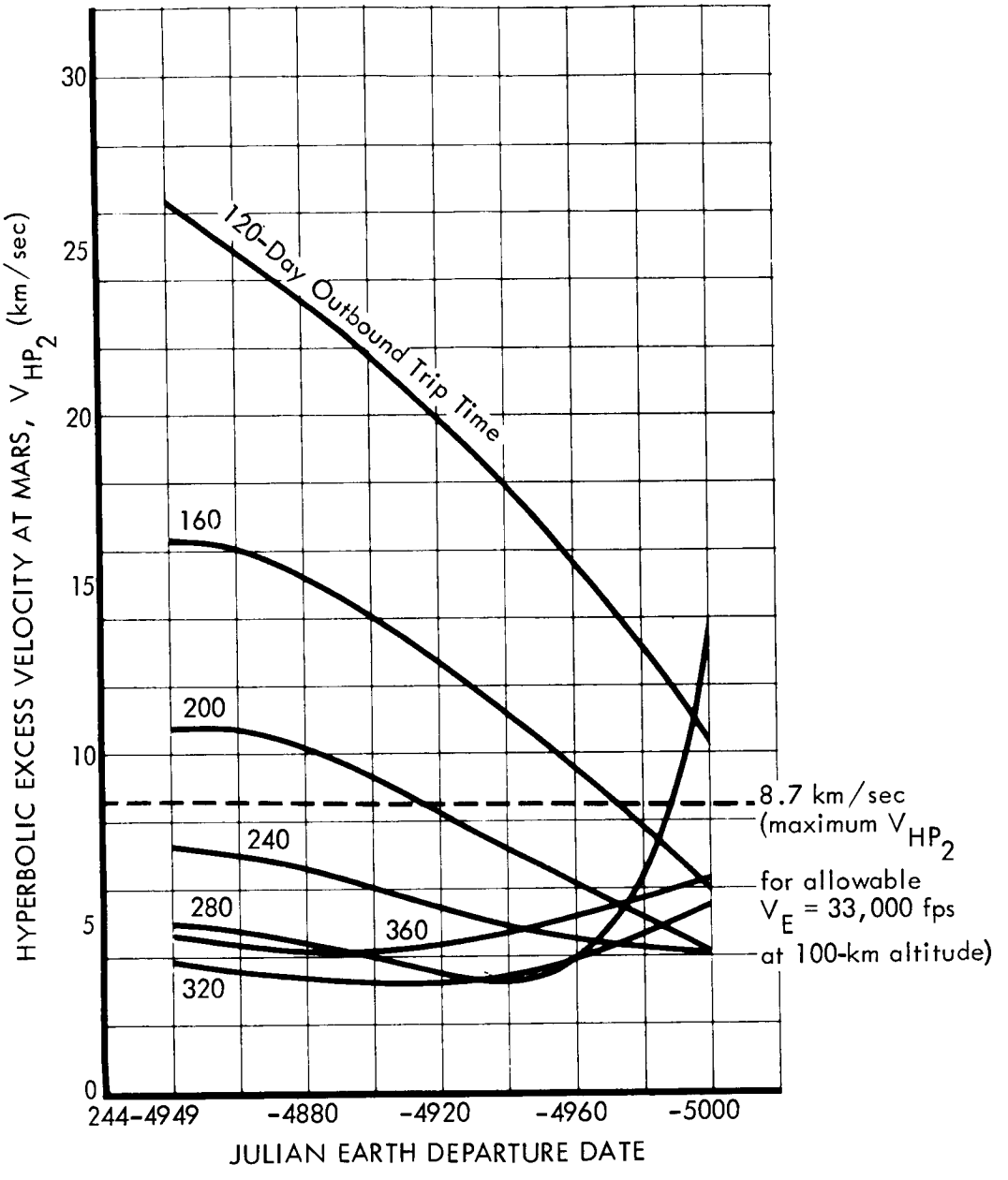


Figure 2.2-10: 1982 MARS OPPOSITION

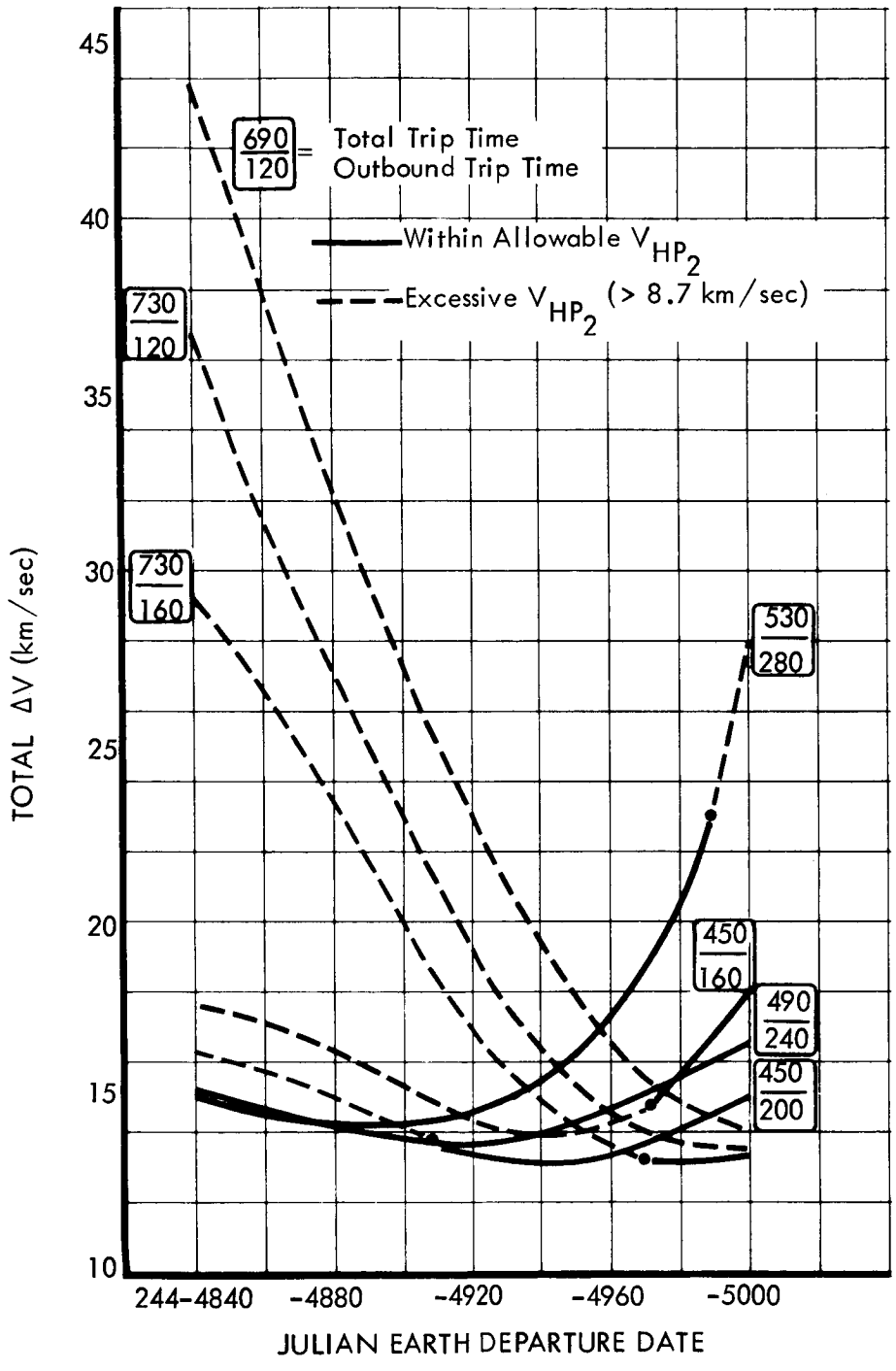


Figure 2.2-11: 1982 MARS OPPOSITION — Aerobraking

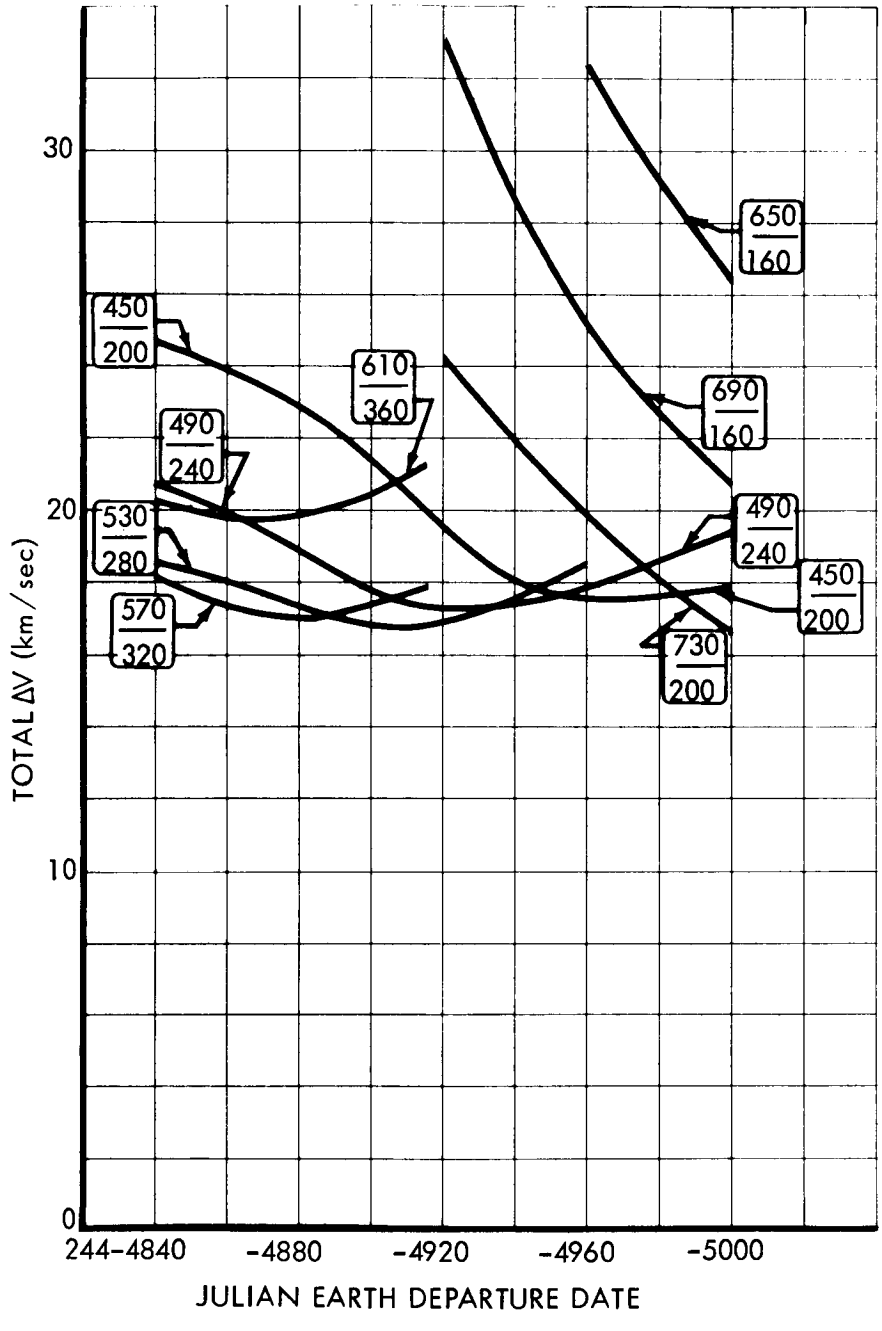


Figure 2.2-12: 1982 MARS OPPOSITION — All-Propulsive

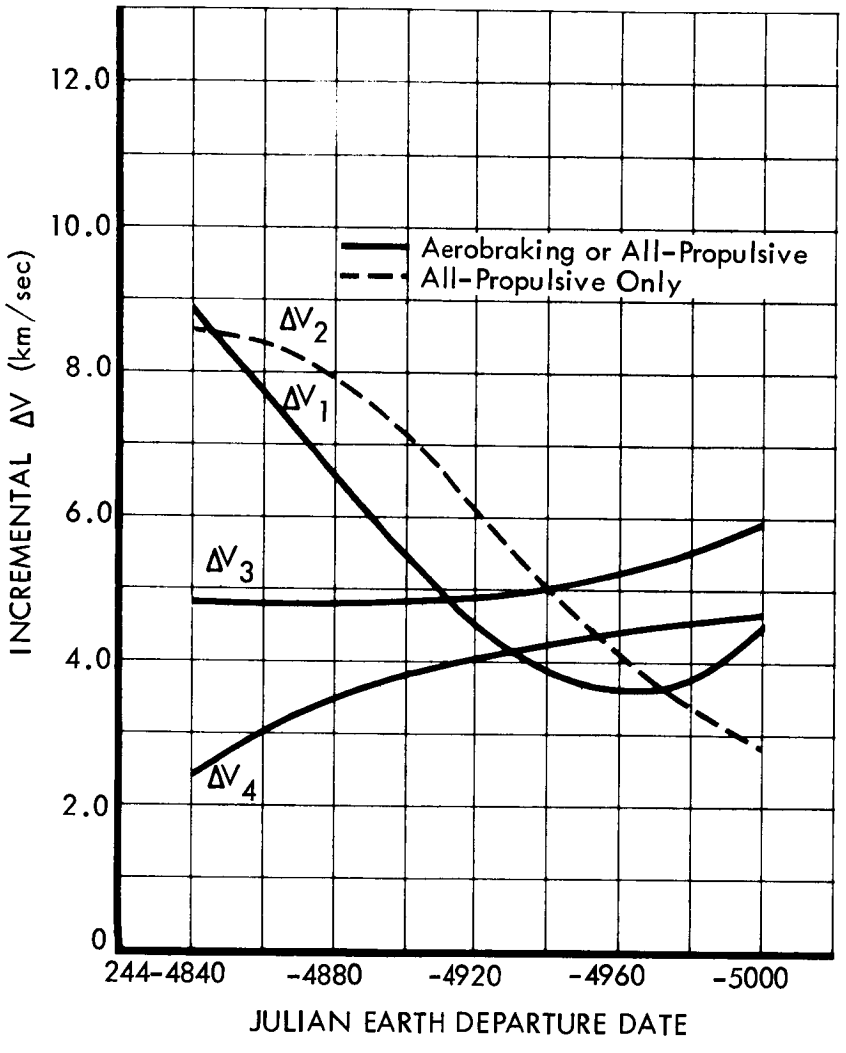


Figure 2.2-13: 1982 MARS OPPOSITION — 450/200 Mission

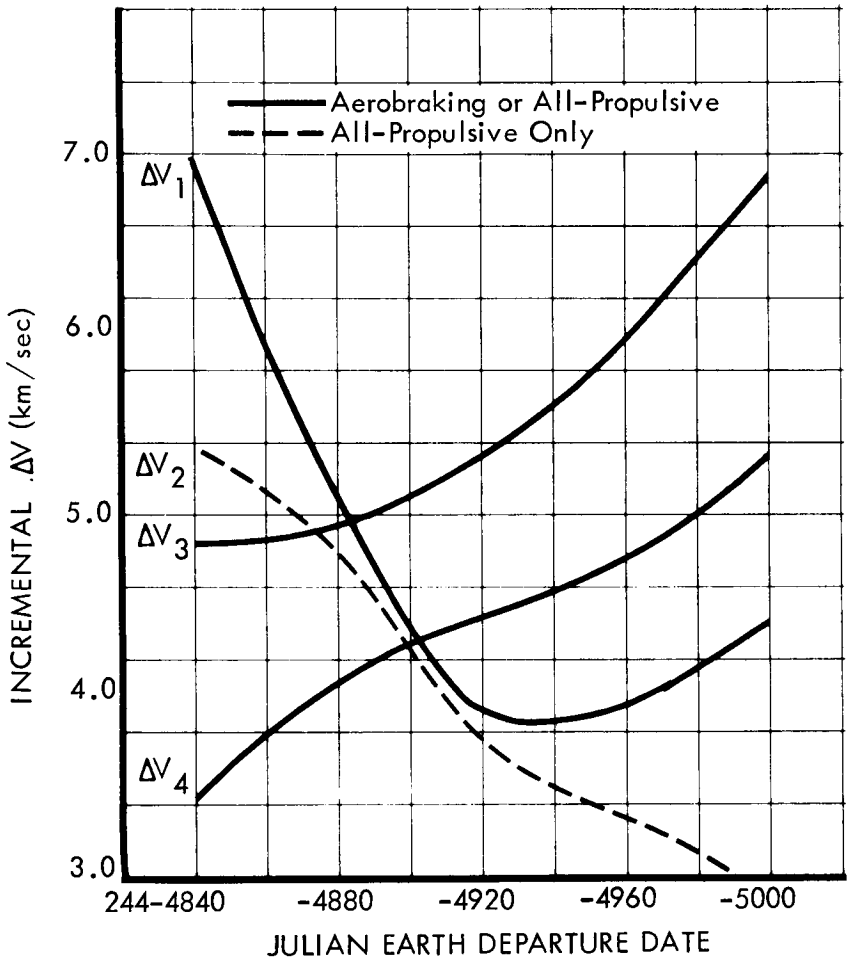


Figure 2.2-14: 1982 MARS OPPOSITION — 490/240 Mission

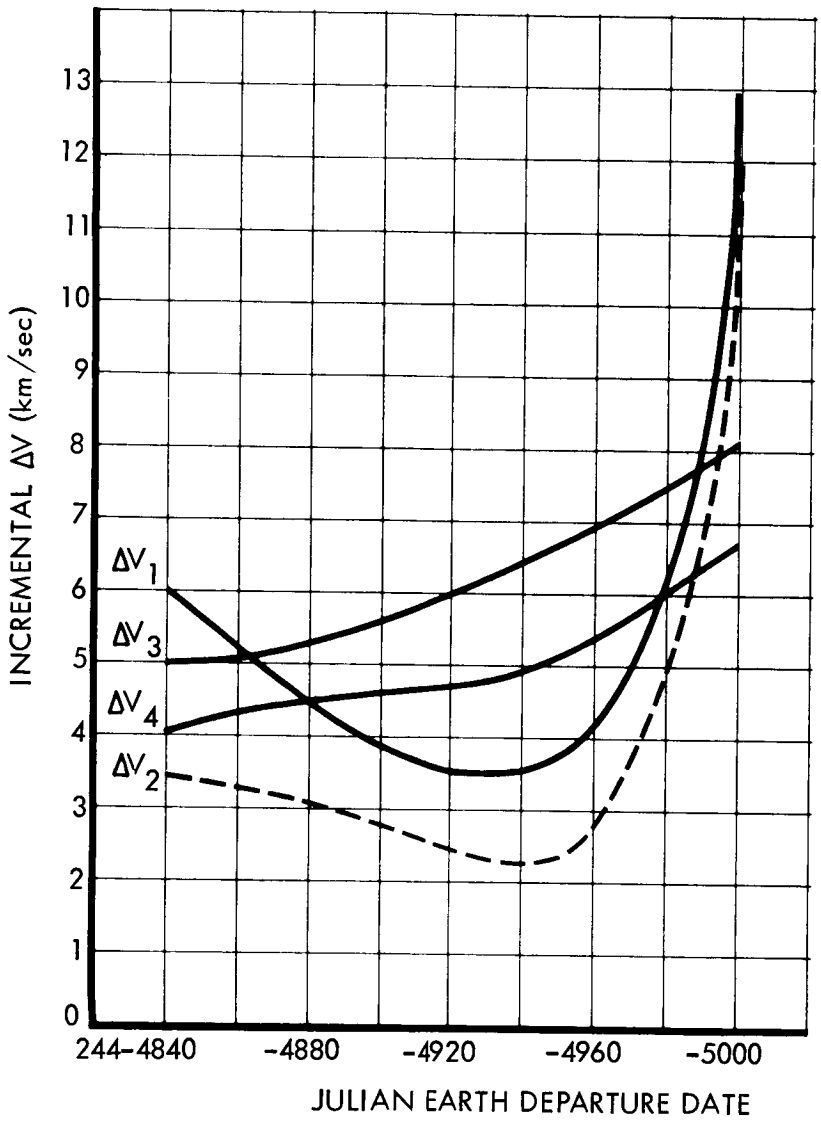


Figure 2.2-15: 1982 MARS OPPOSITION — All-Propulsive — 530/280 Mission

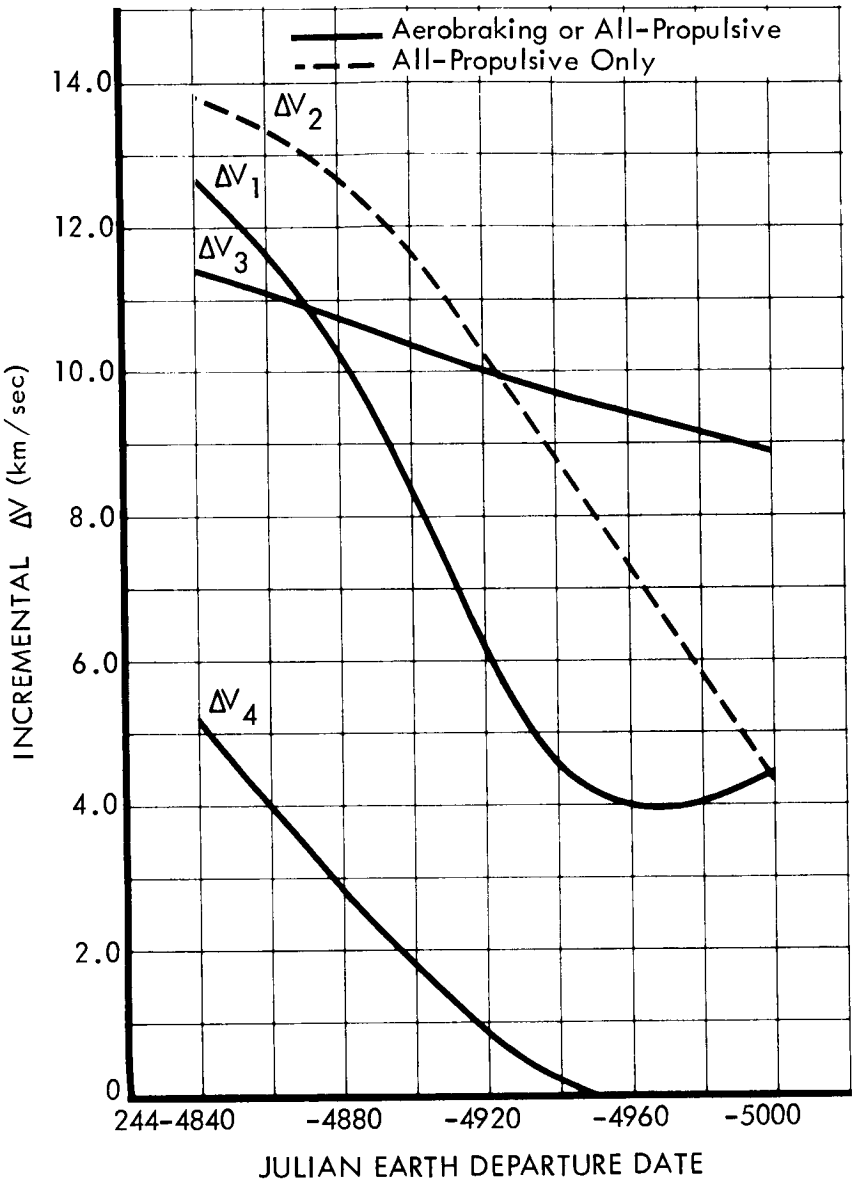


Figure 2.2-16: 1982 MARS OPPOSITION — 730/160 Mission

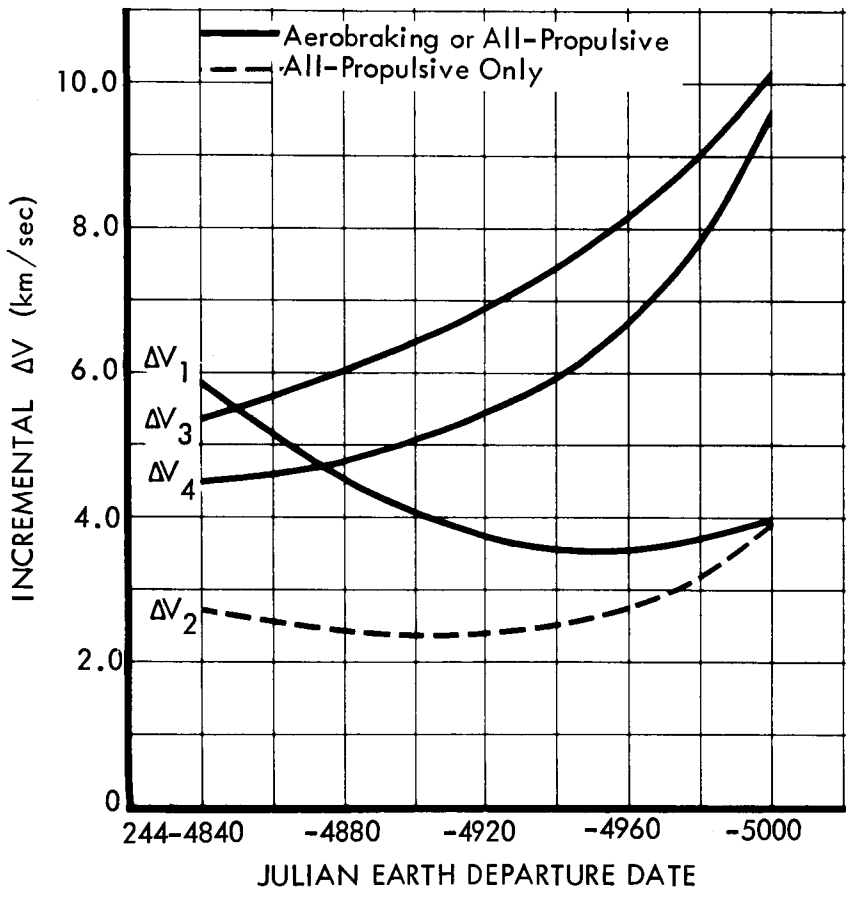


Figure 2.2-17: 1982 MARS OPPOSITION — 570/320 Mission

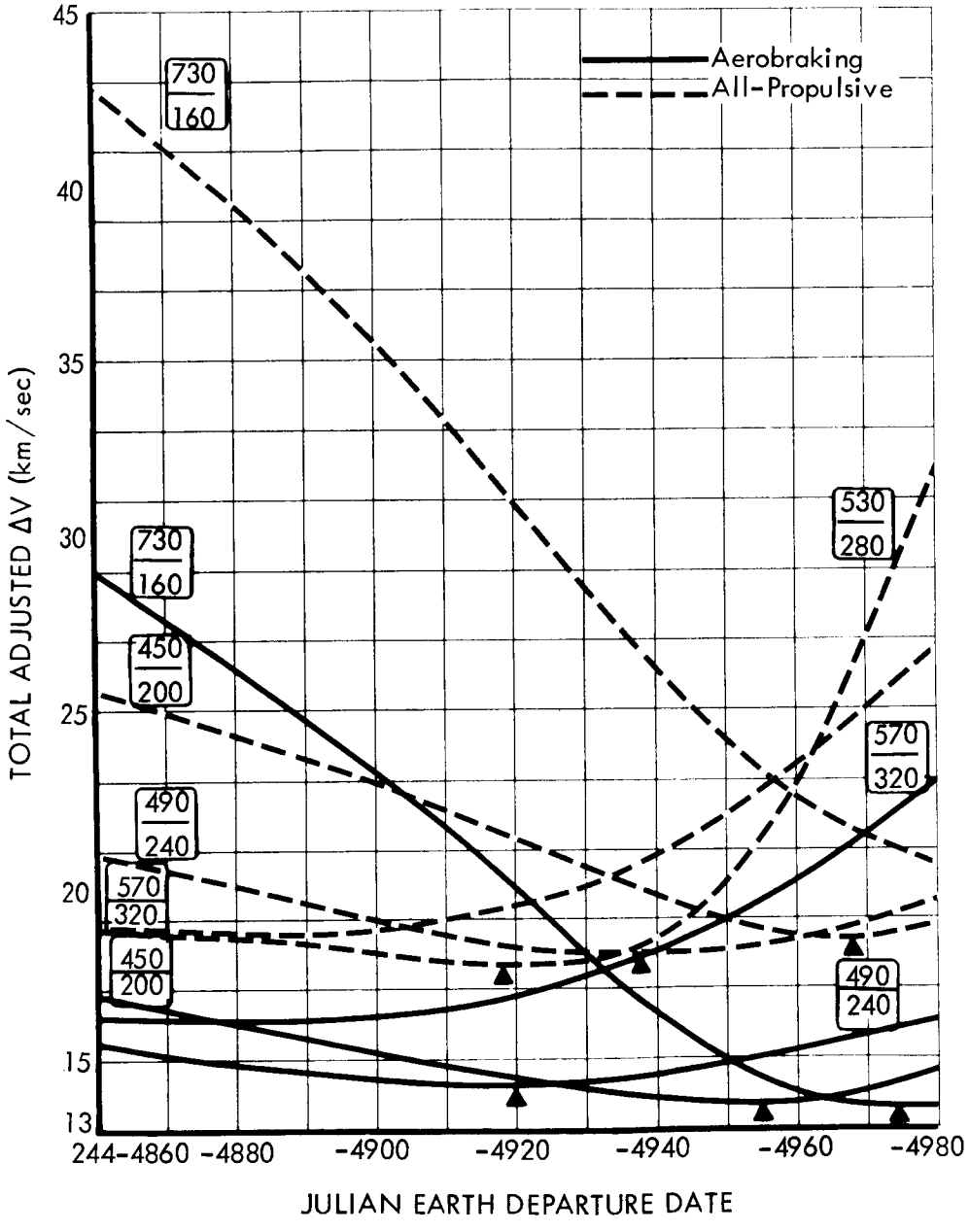


Figure 2.2-18: 1982 MARS OPPOSITION — 30-Day Launch Window

In this chart both the aerobraker missions ($\Delta V_2 = 0$) and the all-propulsive missions were plotted. From this plot the apparent minimum total adjusted ΔV aerobraker missions are the 450/200* and 730/160-day missions with departure dates of J.D. 244-4955 and J.D. 244-4973, respectively, both within the allowable V_{HP_2} range as indicated in Figures 2.2-10 and 2.2-11. Because of shorter total trip time, the 450/200-day mission appears more desirable. The next closest was the 490/240-day mission with a departure date of J.D. 244-4920. For the all-propulsive mission, the 530/280-day mission still appears most desirable with a departure date of J.D. 244-4919. Note that the most desirable missions still remain the same as those that would have been selected on the basis of minimum total ΔV, but the departure dates have shifted to 13 and 15 days later, respectively, for the all-propulsive and aerobraker missions.

7) The mission selections were checked against those that would be chosen on the basis of IMIEO. The lowest energy requirement missions for the aerobraker, the 450/200 and 490/240, were run through the space vehicle design computer program for varying departure dates using a nuclear-aerobraking-chemical (NAC) configuration. Similarly, IMIEO's were calculated for the all-propulsive 530/280-day mission using a nuclear-nuclear-nuclear (NNN) configuration. The NAC and NNN plots are shown in Figures 2.2-19 and 2.2-20, respectively. The mission selection remains the same for the NAC (i.e., 450/200-day mission), and the indicated desired departure dates for the NAC and NNN become J.D. 244-4960 and J.D. 244-4927, respectively. Figure 2.2-20 shows that the mission previously selected on the basis of adjusted ΔV (i.e., departure date of 244-4955) has an IMIEO only 1% different than that of the minimum IMIEO mission.

8) With the data developed in the preceding steps, comparisons can be made between the aerobraker IMIEO's calculated for the all-propulsive type trajectories and those selected specifically for the aerobraker missions. Table 2.2-1 illustrates this comparison for the 1982 Mars opposition mission. The NAC IMIEO calculated using the newly selected all-propulsive trajectories is about 44% greater than the NAC IMIEO calculated using the trajectories newly selected specifically for the aerobraker mission. A new all-propulsive trajectory selection was made in this analysis to ensure consistency in the comparisons since the incremental ΔV inputs to the design program were adjusted ΔV's to accommodate the 30-day Earth departure launch window from a mission energy standpoint. It can be noted that the initial and newly selected all-propulsive mission requirements and NNN IMIEO's for the 1982 Mars opposition mission differ insignificantly. The optimized aerobraker NAC IMIEO for this mission turns out to be practically the same as the NNN all-propulsive IMIEO.

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*Total trip time/outbound trip time

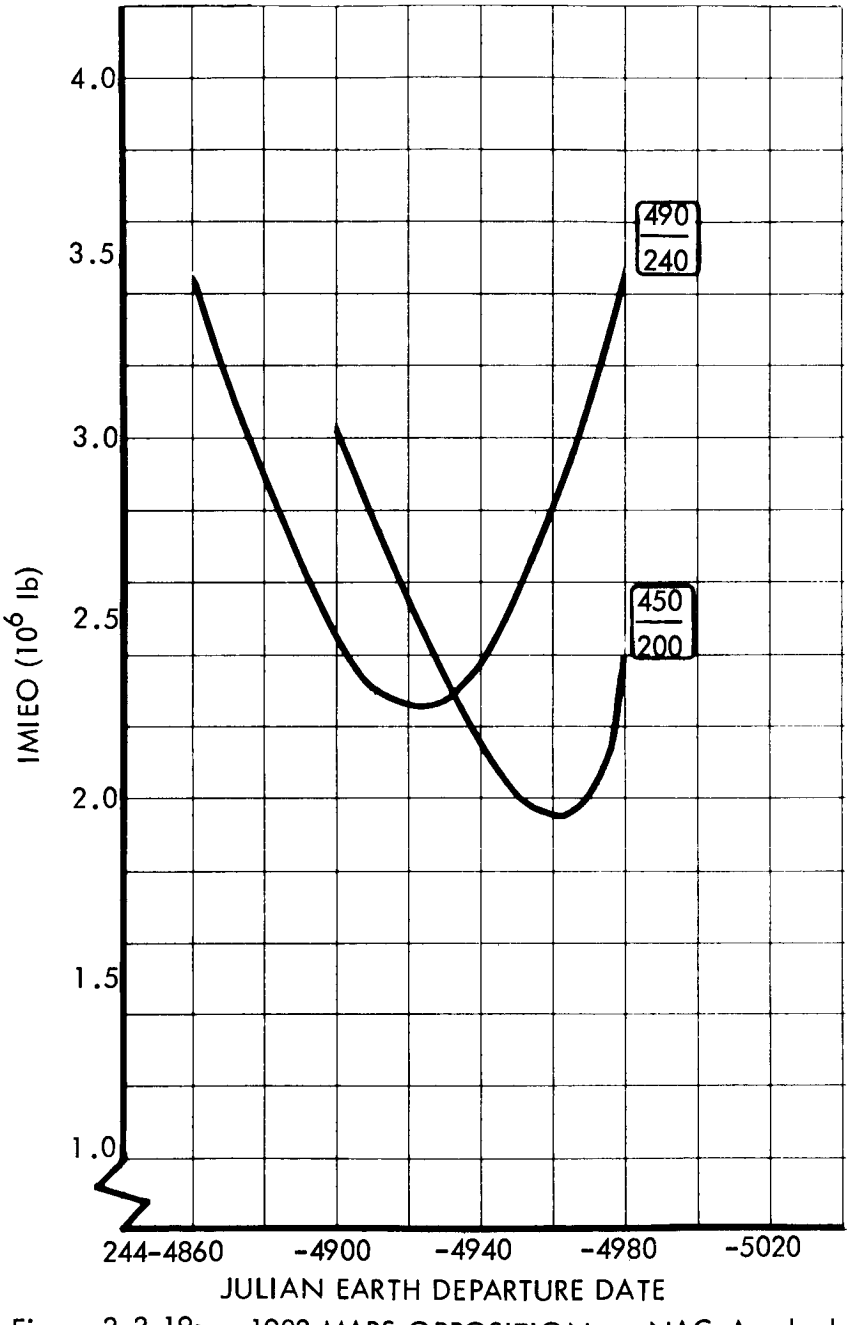


Figure 2.2-19: 1982 MARS OPPOSITION — NAC-Aerobraker

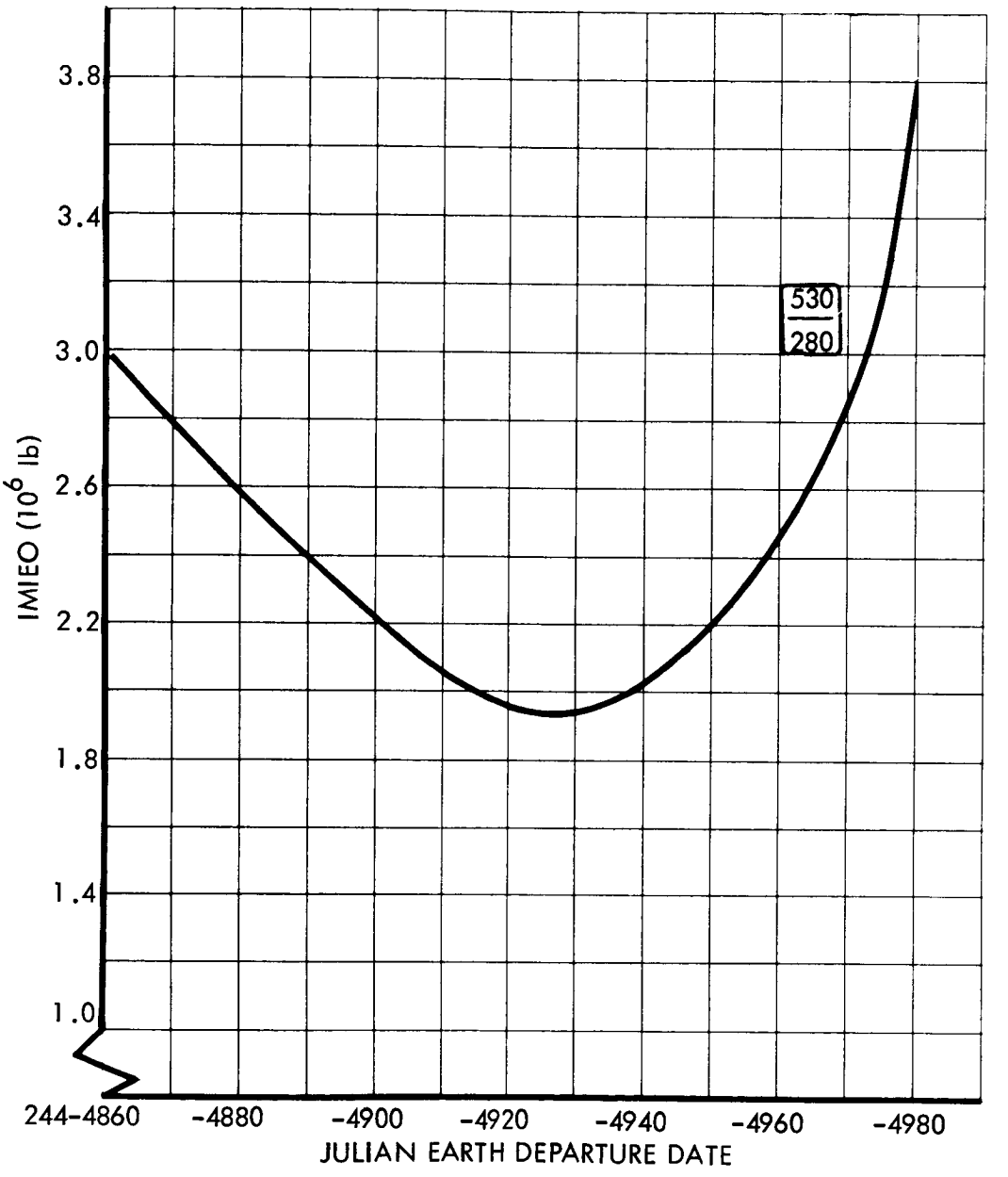


Figure 2.2-20: 1982 MARS OPPOSITION — NNN-All Propulsive

Table 2.2-1: 1982 MARS OPPOSITION AEROBRAKER MISSION SELECTION COMPARISON

	Julian Earth Depart Date	ΔV Requirements				Trip Time		Calculated IMIEO's	
		ΔV_1 (km/sec)	ΔV_2 (km/sec)	ΔV_3 (km/sec)	*ΔV_4 (km/sec)	Out-Bound (days)	Total (days)	NCA (10^6 lb)	NNN (10^6 lb)
Initially Selected Mission (All-Propulsive)	244-4900	3.99	2.57	5.81	4.64	290	540	2.27	1.94
Newly Selected Mission (All-Propulsive)	244-4927	3.70	2.62	6.47	4.97	280	530	2.80	1.93
Newly Selected Mission (Aerobraker)	244-4960	3.75	0** (4.80)	5.47	4.55	200	450	1.95	

*Based on allowable Earth entry velocity of 13,700 m/sec. For IMIEO calculations, an allowable V_E of 18,300 m/sec was assumed and $\Delta V_4 = 0$.

**Number in brackets is the equivalent ΔV that must be taken out by aerobraking.

This constitutes the general approach used in this analysis for each of the missions. The 1982 swingby mission required a slight variation in the approach for matching the outbound and inbound legs where the inbound leg had a variety of combinations of Mars-to-Venus and Venus-to-Earth trip times. From the ΔV plots of different combinations of inbound legs, particular combinations which provided minimum ΔV were selected. For each of those combinations adjusted, ΔV plots were made as a function of Mars departure date. Once the departure dates, which offered minimum adjusted ΔV requirements, were identified, the corresponding adjusted increments were added to adjusted incremental ΔV requirements for different outbound legs. The total adjusted ΔV was then plotted against outbound leg trip time and the minimum outbound leg trip time for the different combinations of inbound legs was determined. IMIEO's were calculated for the minimum adjusted ΔV trajectory combinations and the minimum IMIEO swingby trajectories were selected. For more sophisticated analyses, the trajectories should be optimized for each day of the desired launch window. However, the adjusted ΔV method, using fixed return trip times over the entire launch window, allows a reasonable approximation of the swingby mission requirements. The charts used in this analysis for the other five missions under consideration are found in Appendixes A-4 through A-8.

A summary comparison chart for all six missions is presented in Table 2.2-2. For each mission class, the first mission shown "Initial (All-Propulsive)," is the mission initially selected as described in Sections 2.1.1 and 2.1.3 for representative requirements. The ΔV's shown represent the increments required for that particular departure date which are not necessarily the maximum required across a 30-day launch window even though a 10% deviation from minimum IMIEO was allowed as a margin for a 30-day window. The trajectories selection was made to accommodate an all-propulsive vehicle. All IMIEO's shown in the table were calculated via the vehicle design program. The second mission shown in each class "New (All-Propulsive)," is the mission selected in this analysis for the all-propulsive application also using minimum IMIEO as the selection criteria, but with the incremental ΔV's adjusted to more realistically reflect the actual trajectory ΔV requirements over the 30-day window. Additional departure penalties to ΔV_1 are investigated and described in Section 2.2.2. The third mission of each class "New (Aerobraker)," is the mission selected in this analysis which is most advantageous for the aerobraker from an IMIEO standpoint. In all six missions, the aerobraker IMIEO's are penalized by having to fly trajectories selected for all-propulsive vehicles, and the penalty is generally greater for the higher energy missions. However the range of IMIEO's for the optimized aerobraker trajectories is not significantly different than the range for the initial aerobraker IMIEO's using optimum propulsive trajectories. Therefore, the system trade studies were not significantly biased.

It is also interesting to note that the aerobrakers (NAC) were 1 and 14% higher in IMIEO than the all-propulsive (NNN) vehicles for the 1982 Mars opposition and 1982 Mars with Venus swingby missions, respectively. The aerobraker and all-propulsive vehicles' IMIEO's were the same for the 1986 opposition mission, but the all-propulsive (NNN) vehicles' IMIEO's were 27, 51, and 49% greater than the aerobrakers (NAC) for the 1986 Mars conjunction, 1980 Venus with short stay, and 1980 Venus with long stay missions, respectively.

The importance of considering launch window variations in each incremental ΔV was also demonstrated in this analysis. The "Initial (All-Propulsive)" missions shown in Table 2.2-2, which were selected and used in the IMISCD study, made an arbitrary allowance for launch window by backing up the minimum IMIEO curve by about 10%, resulting in higher than nominal ΔV requirements. However, that method does not necessarily guarantee adequate ΔV in each stage to cover the entire launch window. In three missions--1982 Mars opposition, 1986 Mars conjunction, and 1980 Venus with long stay--the NNN IMIEO's calculated for the "Initial (All-Propulsive)" missions and for the "New (All-Propulsive)" missions varied insignificantly. However, in the 1982 Mars with Venus swingby and 1980 Venus with short stay missions, the IMIEO's for the "New (All-Propulsive)" Trajectories, using the adjusted incremental ΔV's, are 23 and 33% higher, respectively, than the "Initial (All-Propulsive)" trajectories which used the 10% IMIEO adjustment for launch window. These IMIEO variations seem to be sufficiently large that consideration of launch window variations in the individual increments of ΔV become important. However, the importance attached to this

Table 2.2-2: AEROBRAKER MISSIONS SELECTION COMPARISON

	JULIAN EARTH DEPART DATE	ΔV REQUIREMENTS (km/sec)				TRIP TIME		CALCULATED IMIEO	
		ΔV_1	ΔV_2	ΔV_3	ΔV_4*	OUT-BOUND (days)	STAY/ TOTAL (days)	NAC ($\times 10^6$ lb)	NNN ($\times 10^6$ lb)
1982 Mars Opposition:									
INITIAL (ALL-PROP)	244-4900	3.99	2.57	5.81	4.64	290	40/540	2.27	1.94
NEW (ALL-PROP)	244-4927	3.70	2.62	6.47	4.97	280	40/530	2.80	1.93
NEW (AEROBRK)	244-4960	3.75	0** (4.80)	5.47	4.55	200	40/450	1.95	
1986 Mars Opposition:									
INITIAL (ALL-PROP)	244-6520	3.65	2.83	4.97	2.55	190	40/480	Not Computed	Not Computed
NEW (ALL-PROP)	244-6541	3.56	2.88	5.75	4.74	180	40/480	2.05	1.67
NEW (AEROBRK)	244-6541	3.75	0** (5.25)	4.90	1.71	140	40/420	1.67	
1986 Mars Conjunction:									
INITIAL (ALL-PROP)	244-6530	3.68	2.47	2.71	0	210	580/1040	1.16	1.43
NEW (ALL-PROP)	244-6548	3.57	2.43	2.77	0	200	590/1050	1.14	1.44
NEW (AEROBRK)	244-6545	3.60	0** (2.50)	2.75	0	200	590/1050	1.13	
1982 Mars/w Venus Swingby:									
INITIAL (ALL-PROP)	244-4910	3.80	2.34	4.55	0.11/0	260	40/600	1.71	1.55
NEW (ALL-PROP)	244-4930	3.63	2.55	4.55	1.25/0	280	40/630	2.20	1.90
NEW (AEROBRK)	244-4928	3.65	0** (2.60)	4.65	1.05/0	280	40/630	2.17	
1980 Venus/w Short Stay:									
INITIAL (ALL-PROP)	244-4310	3.86	4.54	4.07	0.48	130	40/460	1.35	1.59
NEW (ALL-PROP)	244-4314	4.20	5.10	4.70	0.80	140	40/480	1.64	2.12
NEW (AEROBRK)	244-4333	4.25	0** (6.50)	4.20	0.60	100	40/440	1.40	
1980 Venus/w Long Stay:									
INITIAL (ALL-PROP)	244-4320	3.66	4.54	3.40	0	190	430/800	1.18	1.72
NEW (ALL-PROP)	244-4348	3.80	4.45	3.70	0	150	430/760	1.23	1.73
NEW (AEROBRK)	244-4355	3.85	0** (4.65)	3.50	0	150	430/760	1.16	

*Based on allowable earth entry velocity of 45,000 fps. For IMIEO calculations, and allowable V_E of 65,000 fps was assumed and $\Delta V_4 = 0$.

**Number in brackets is the equivalent ΔV that must be taken out by aerobraking.

general result must be tempered, according to the flexibility designed into the propulsion stages. With the inflight propellant transfer between tanks provided in the recommended system, the vehicle has the capability to adjust for individual ΔV variations across the launch window, and the effect on IMIEO is minimized.

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APPENDIX A TO SECTION 2.0

This appendix contains graphs used in the selection of specific trajectories for the study. There are eight sections:

Section	Referenced in Report Section
A-1	2.1.3.1
A-2	2.1.3.3
A-3	2.1.3.4
A-4	2.2.3
A-5	2.2.3
A-6	2.2.3
A-7	2.2.3
A-8	2.2.3

APPENDIX A-1

Opposition-Class Mars Missions and Venus Short Stay Time Missions. Figure 2.1-6 through 2.1-14. Explanation of these curves is given in Section 2.1.3.1.

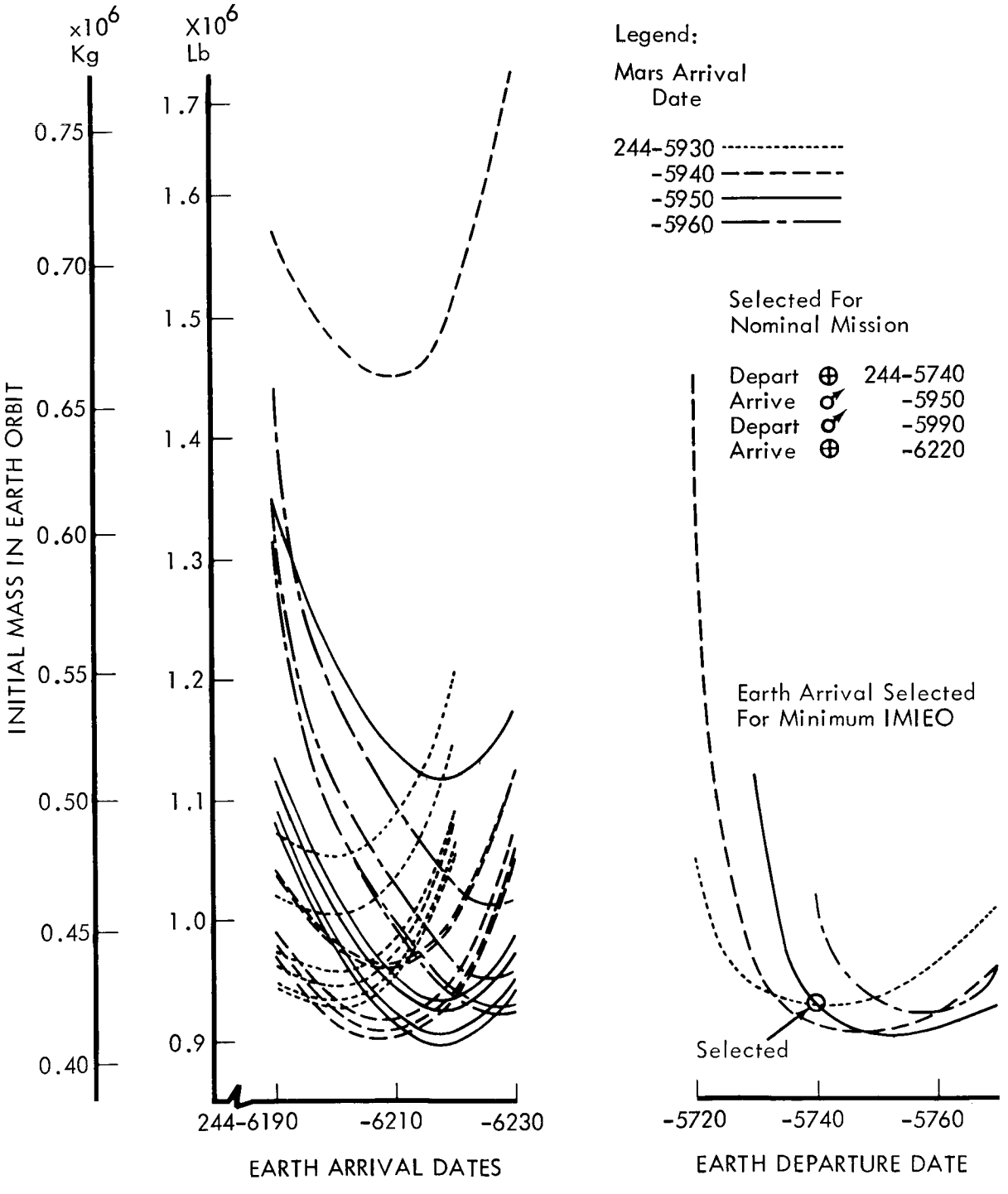


Figure 2.1-6: 1984 OPPOSITION-CLASS TRAJECTORIES

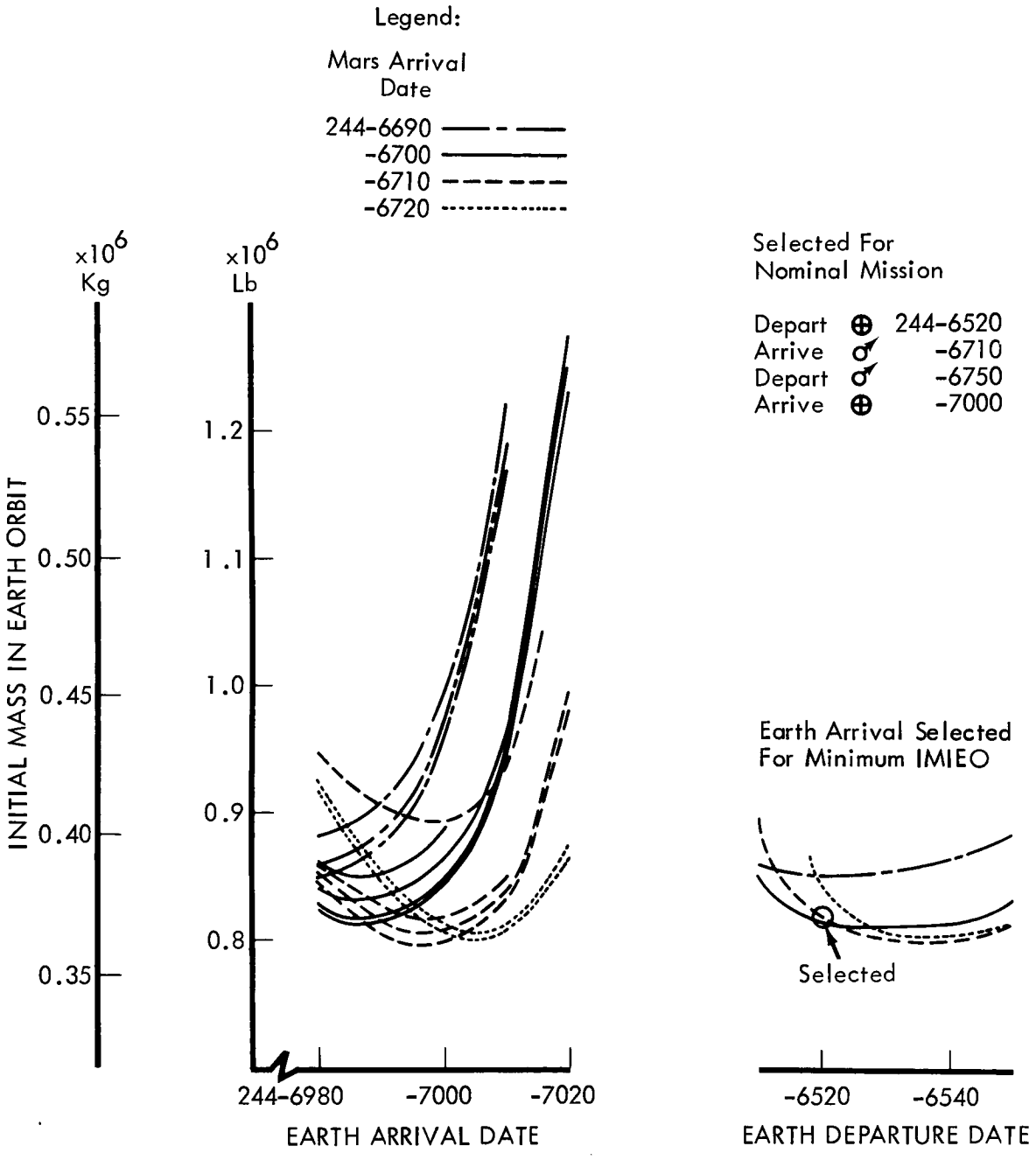


Figure 2.1-7: 1986 MARS OPPOSITION-CLASS TRAJECTORIES

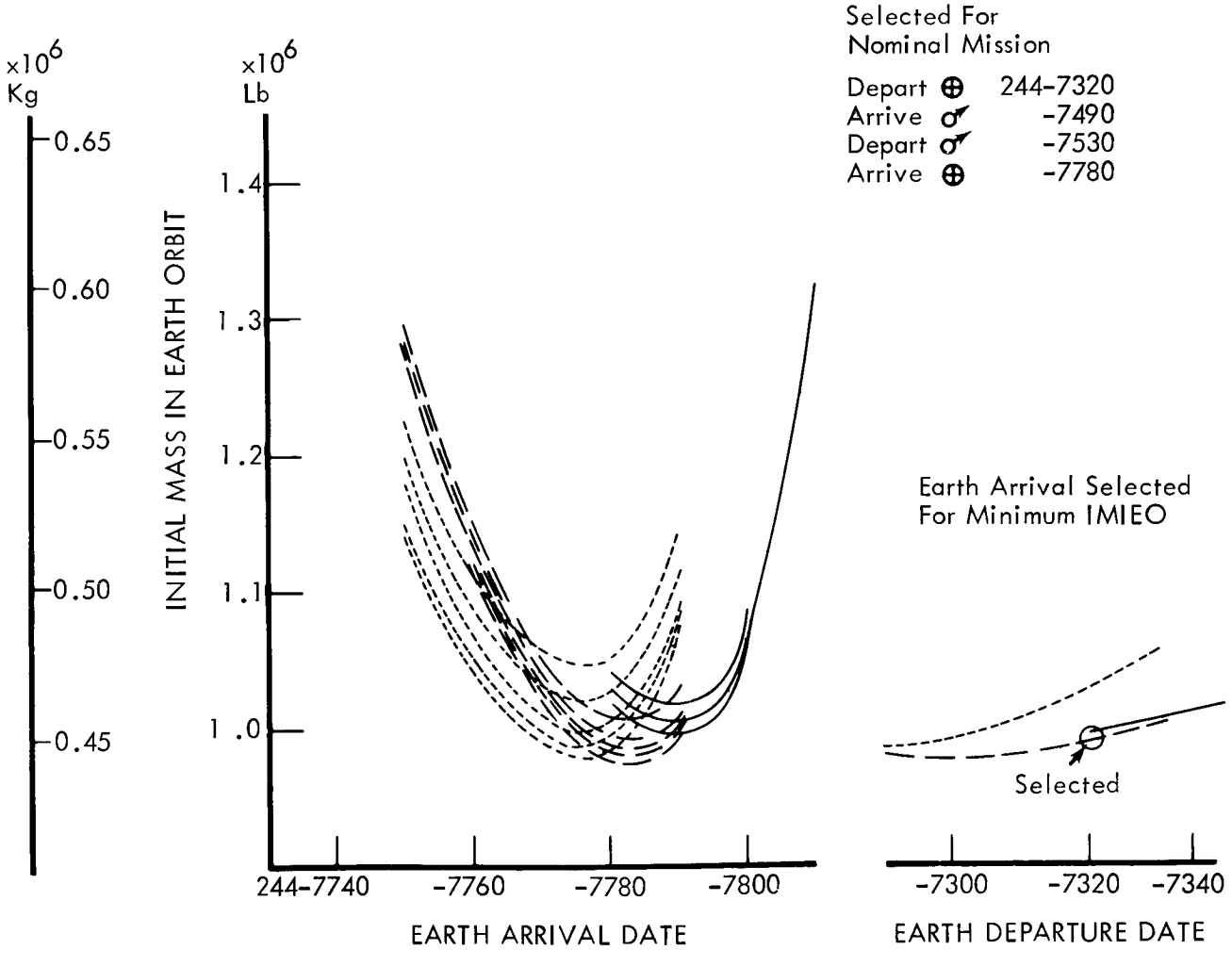


Figure 2.1-8: 1988 MARS OPPOSITION-CLASS TRAJECTORIES

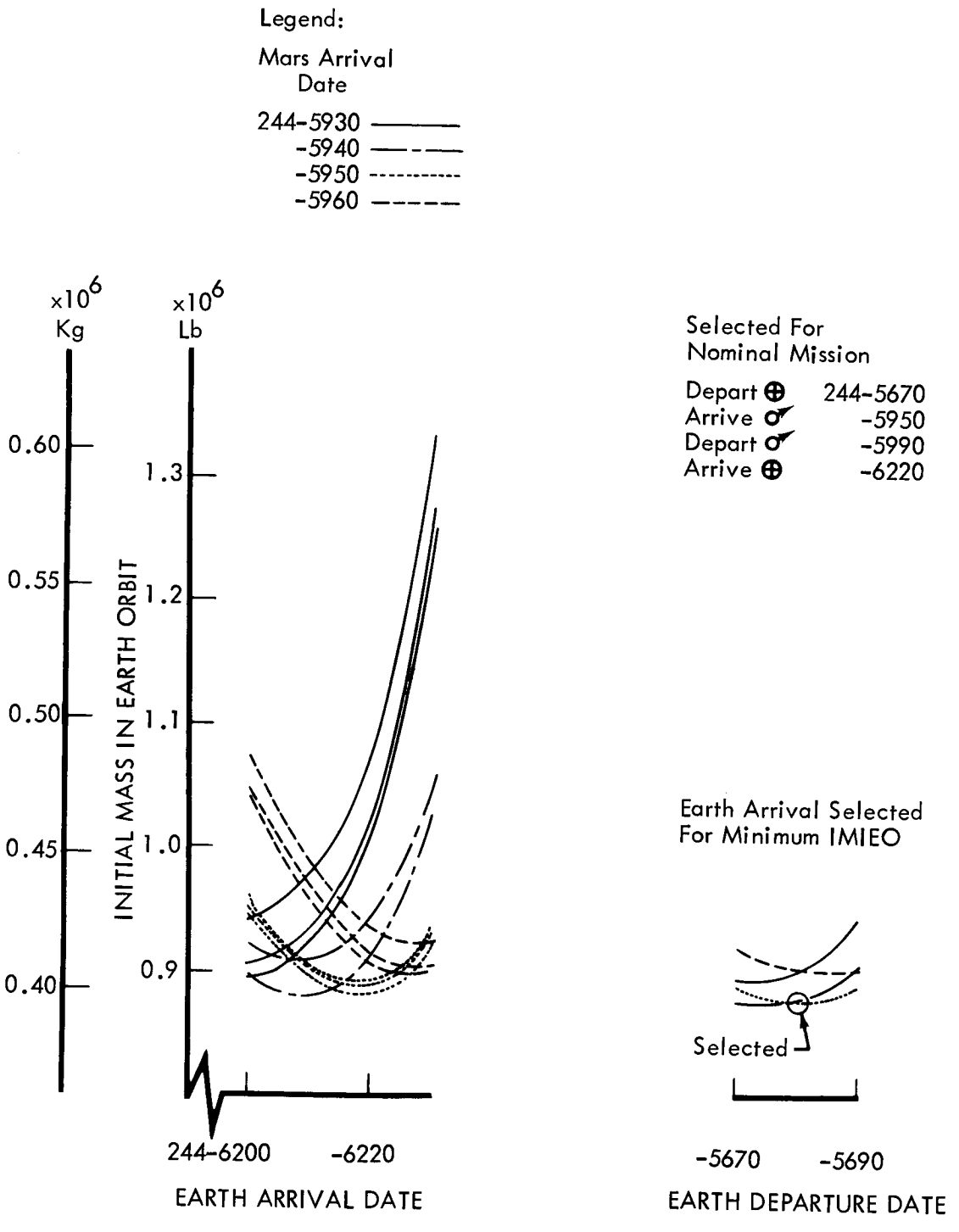


Figure 2.1-9: ALTERNATE 1984 MARS OPPOSITION-CLASS TRAJECTORIES

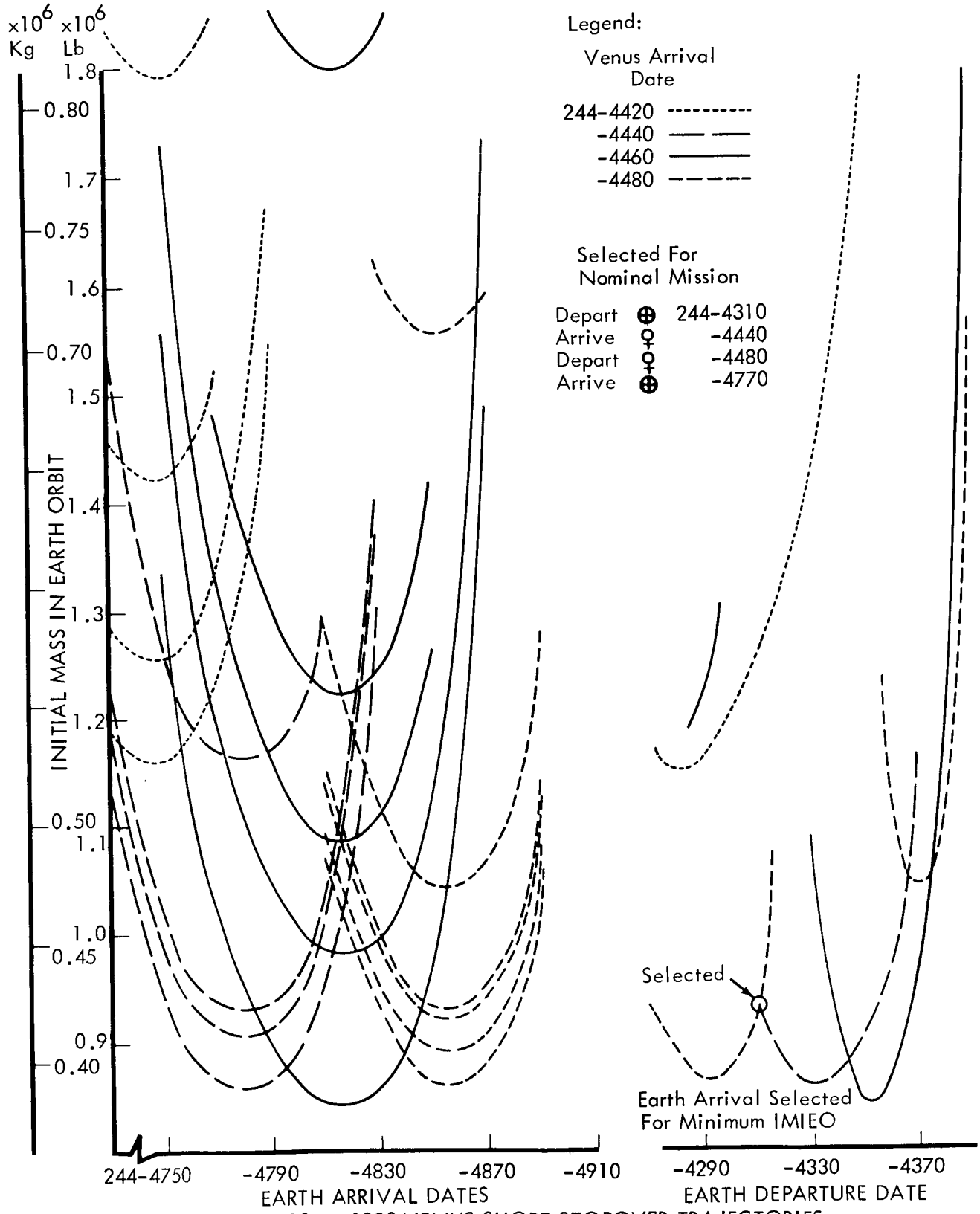


Figure 2.1-10: 1980 VENUS SHORT STOPOVER TRAJECTORIES

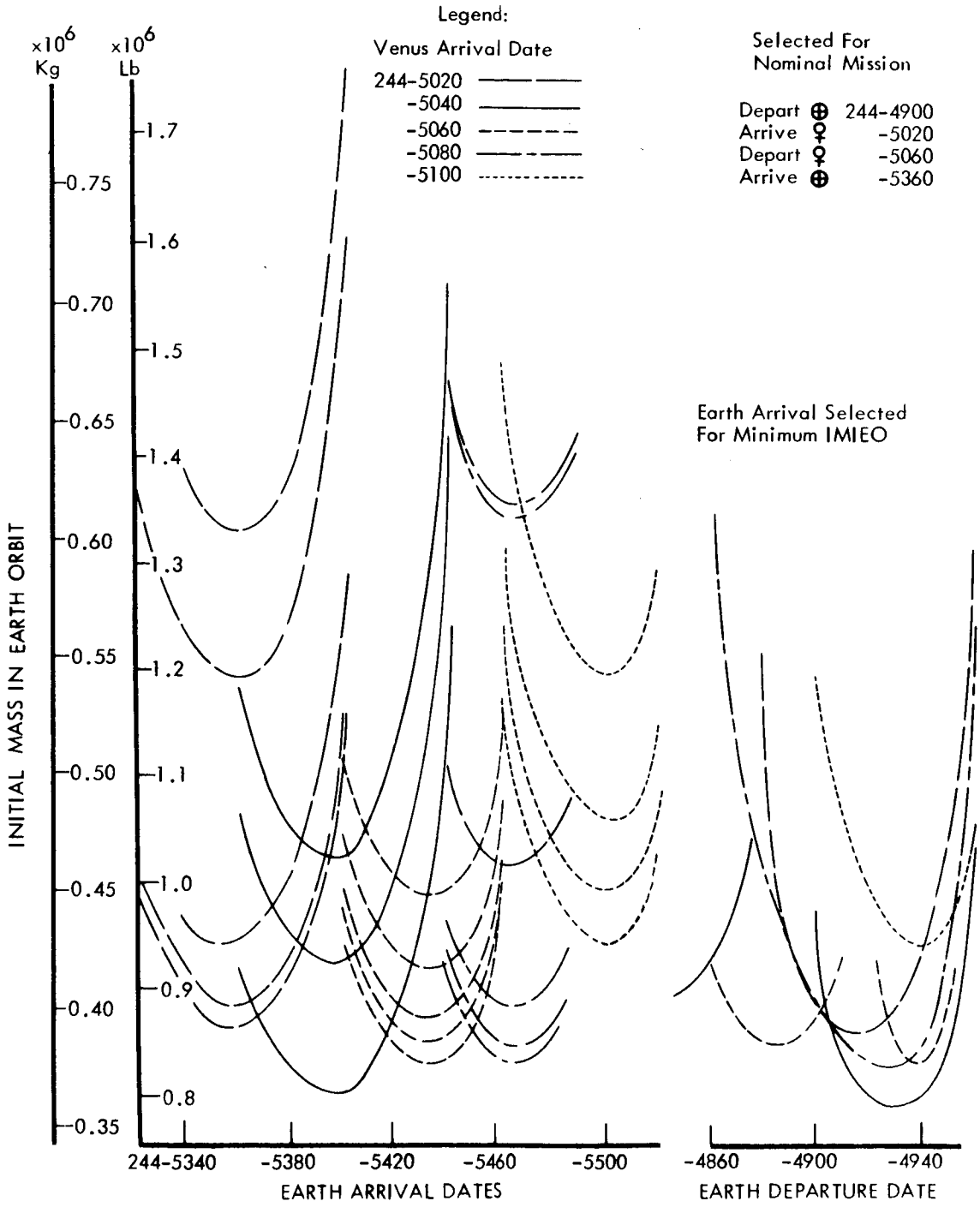


Figure 2.1-11: 1981 VENUS SHORT STOPOVER TRAJECTORIES

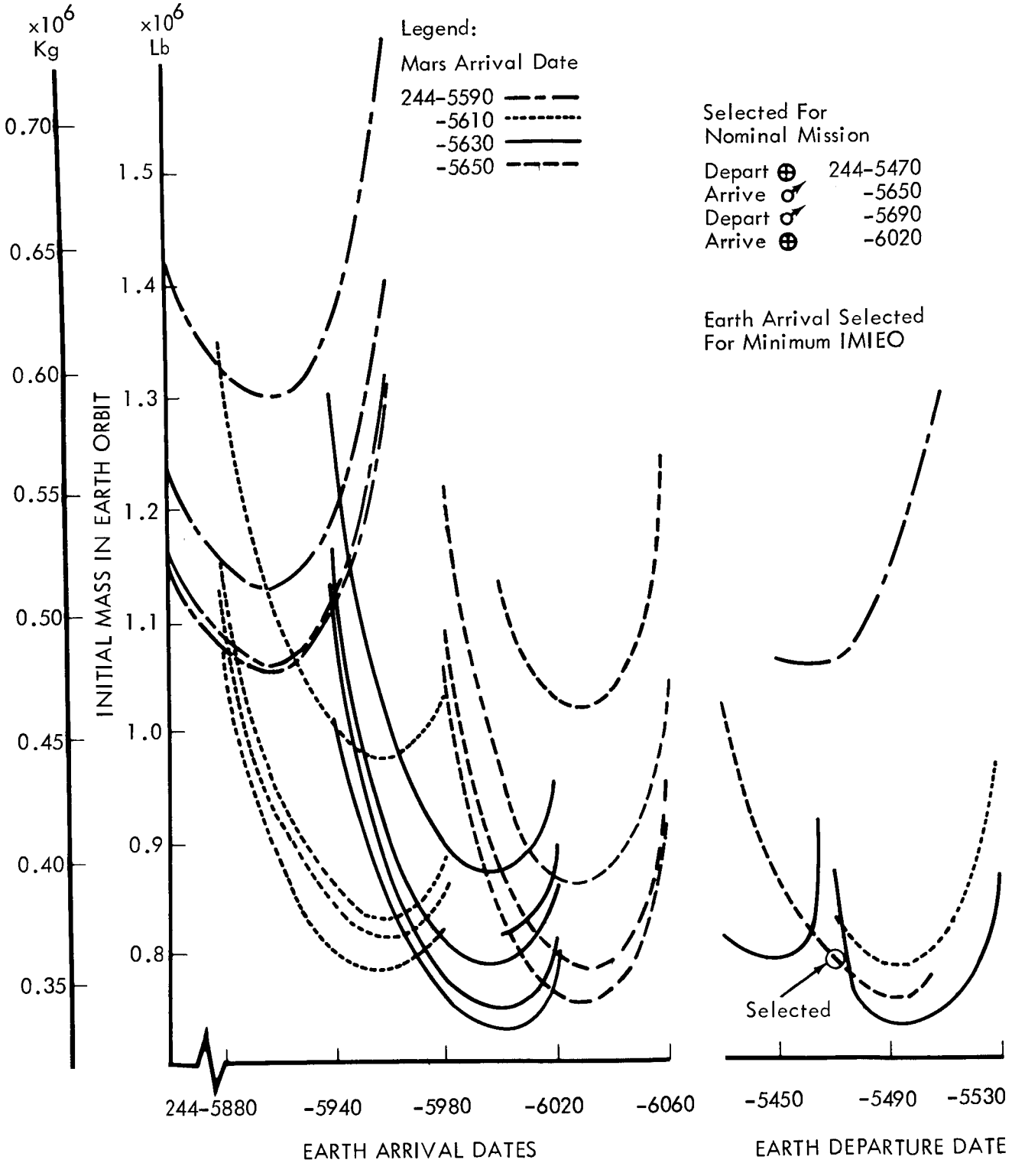


Figure 2.1-12: VENUS SHORT STOPOVER TRAJECTORIES

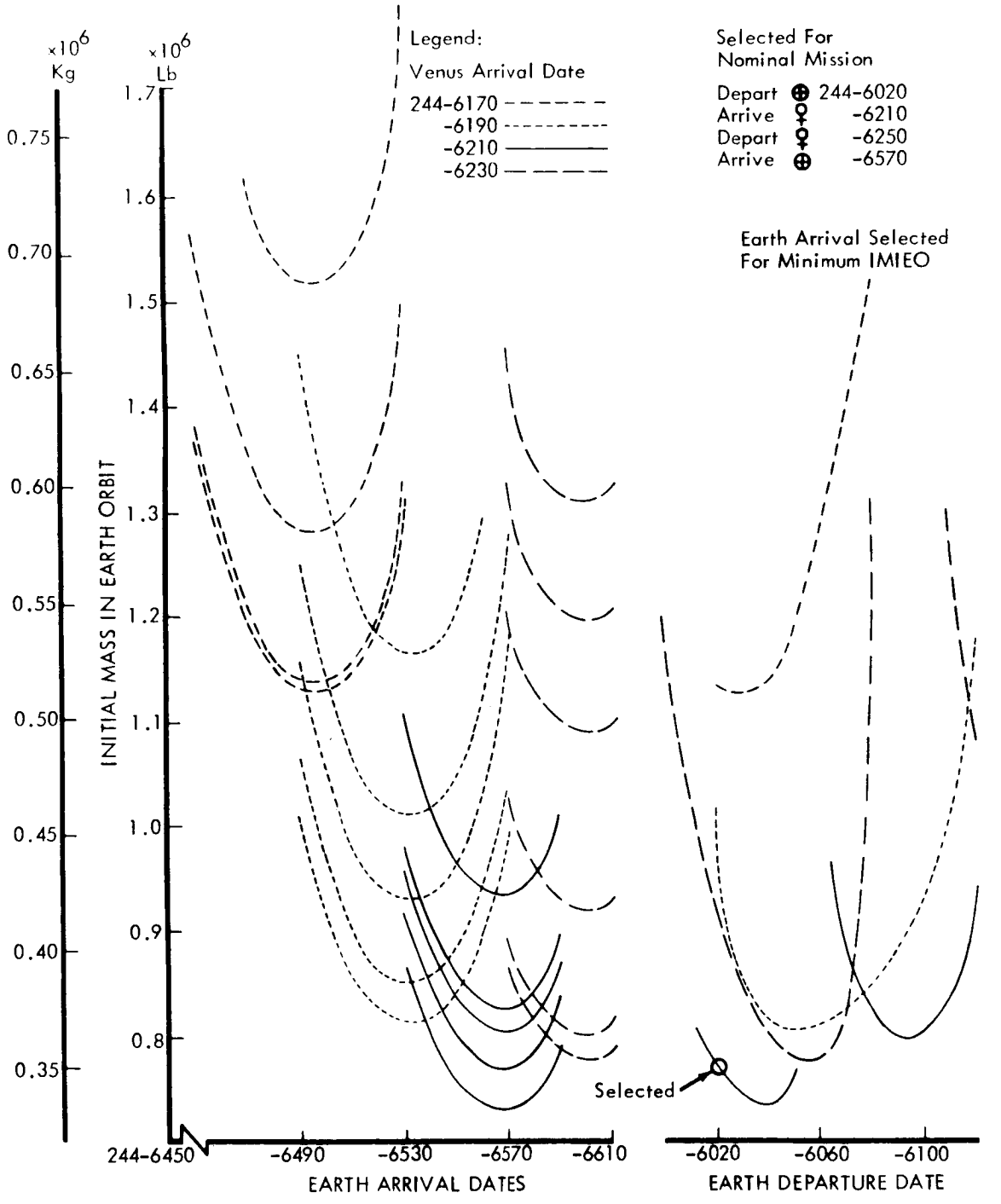


Figure 2.1-13: 1985 VENUS SHORT STOPOVER TRAJECTORIES

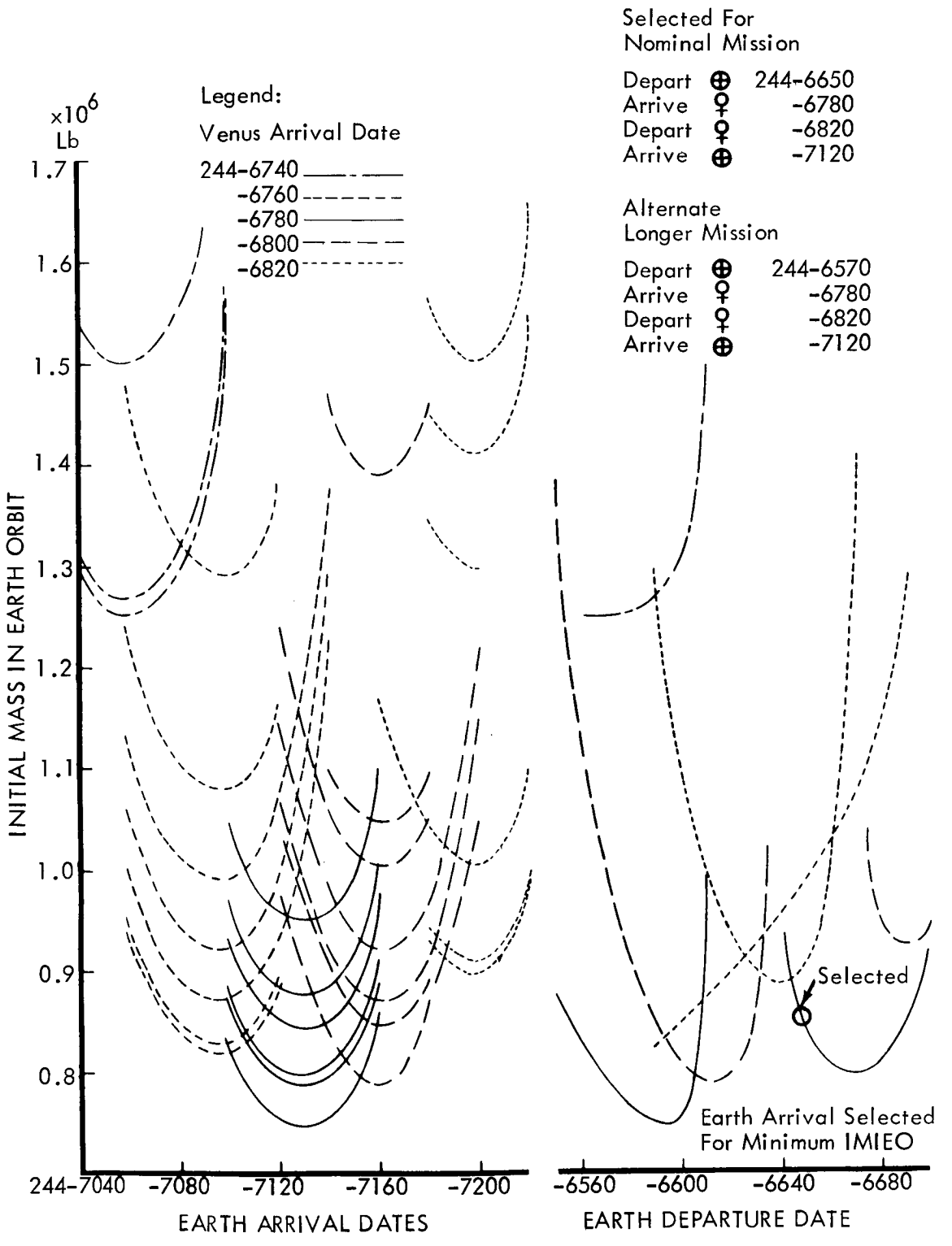


Figure 2.1-14: VENUS SHORT STOPOVER TRAJECTORIES

APPENDIX A-2

Mars Conjunction Class Missions and Venus Long Stay Time Missions. Figures 2.1-17 through 2.1-20. Explanation of these curves is given in Section 2.1.3.3.

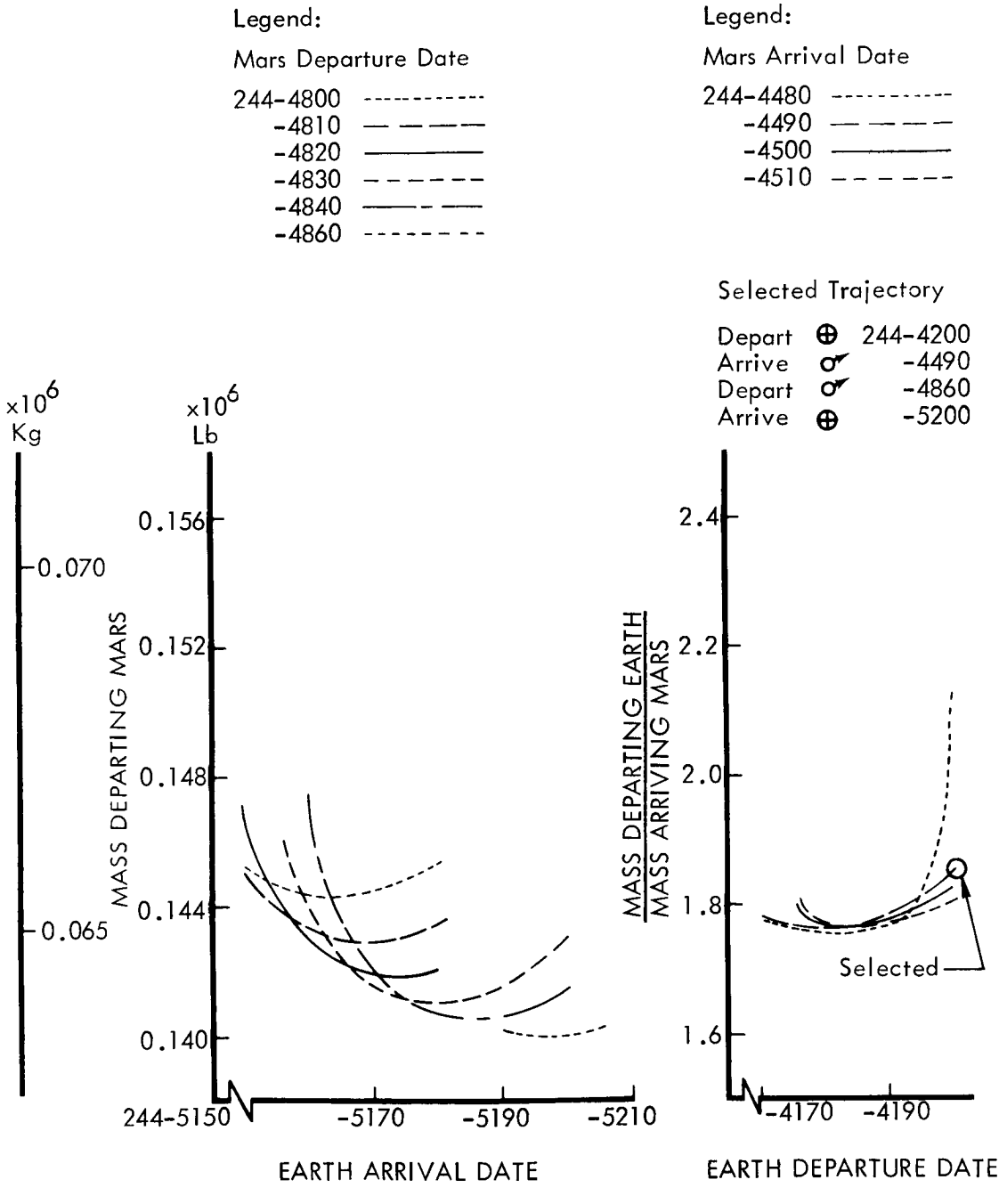


Figure 2.1-17: 1980 MARS CONJUNCTION-CLASS TRAJECTORIES

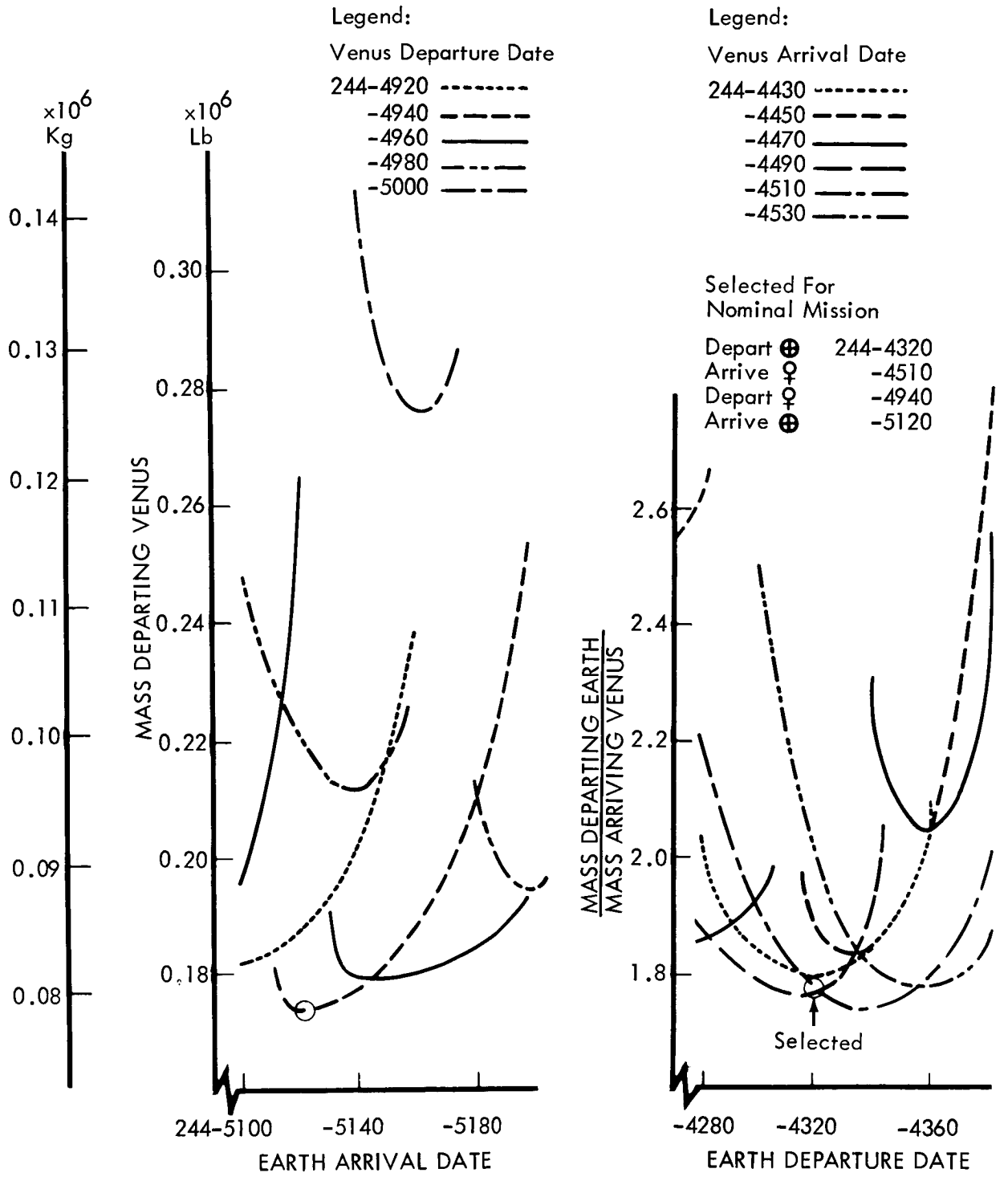


Figure 2.1-18: 1980 VENUS LONG STOPOVER TRAJECTORIES

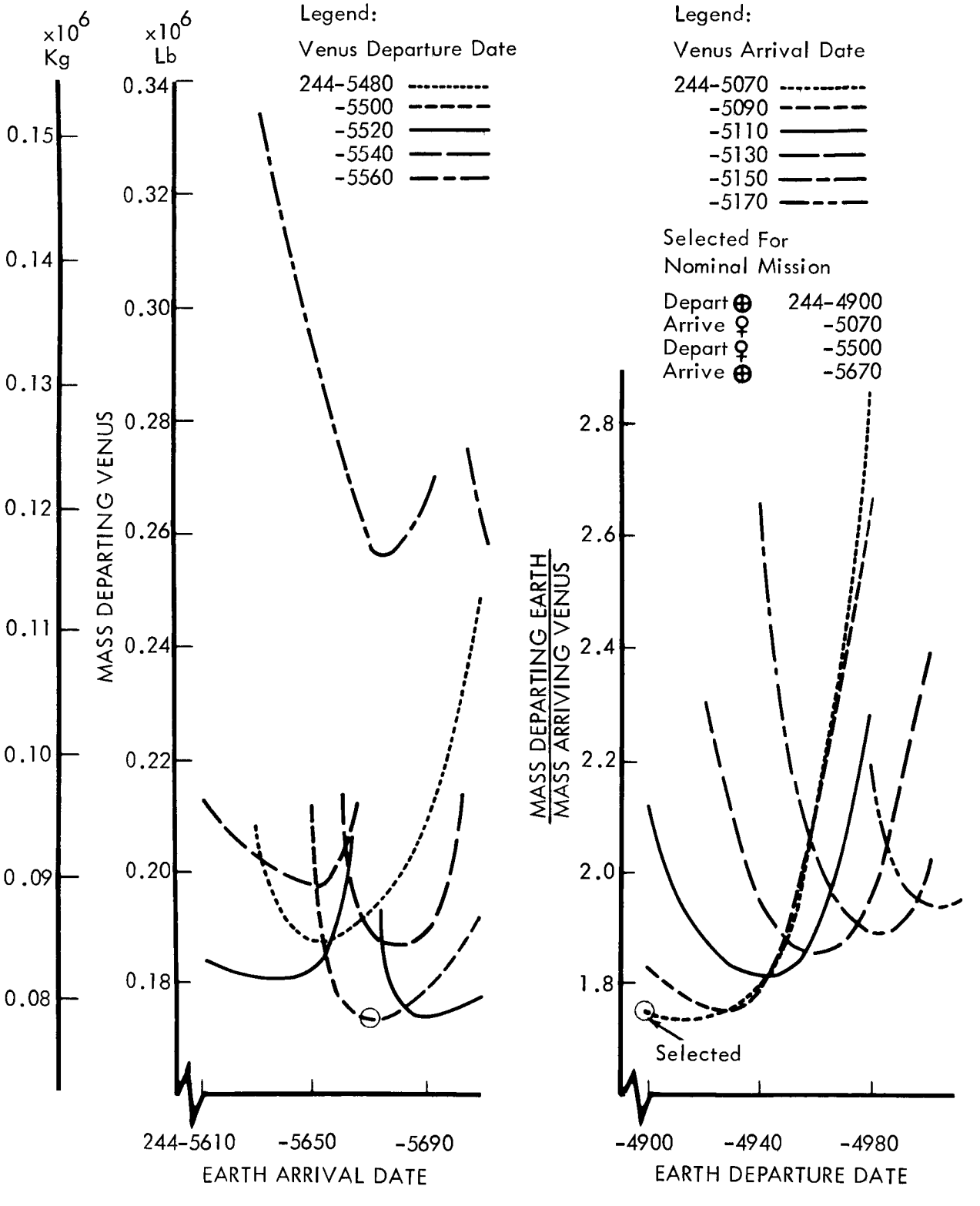


Figure 2.1-19: 1981 VENUS LONG STOPOVER TRAJECTORIES

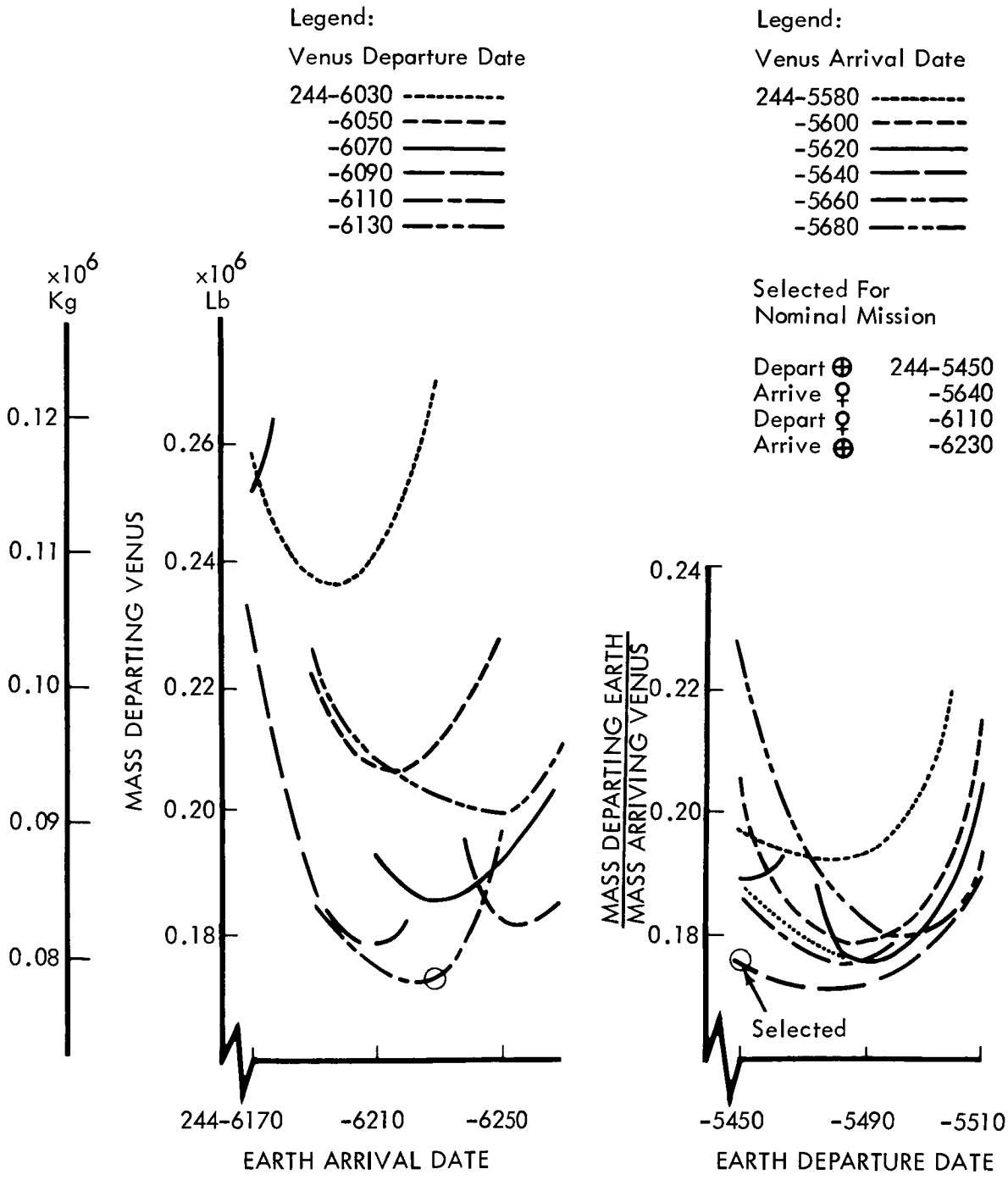


Figure 2.1-20: 1983 VENUS LONG STOPOVER TRAJECTORIES

APPENDIX A-3

Mars Trajectories with Venus Swingby.
Figures 2.1-24 through 2.1-28. Explanation of these curves is given in Section 2.1.3.4.

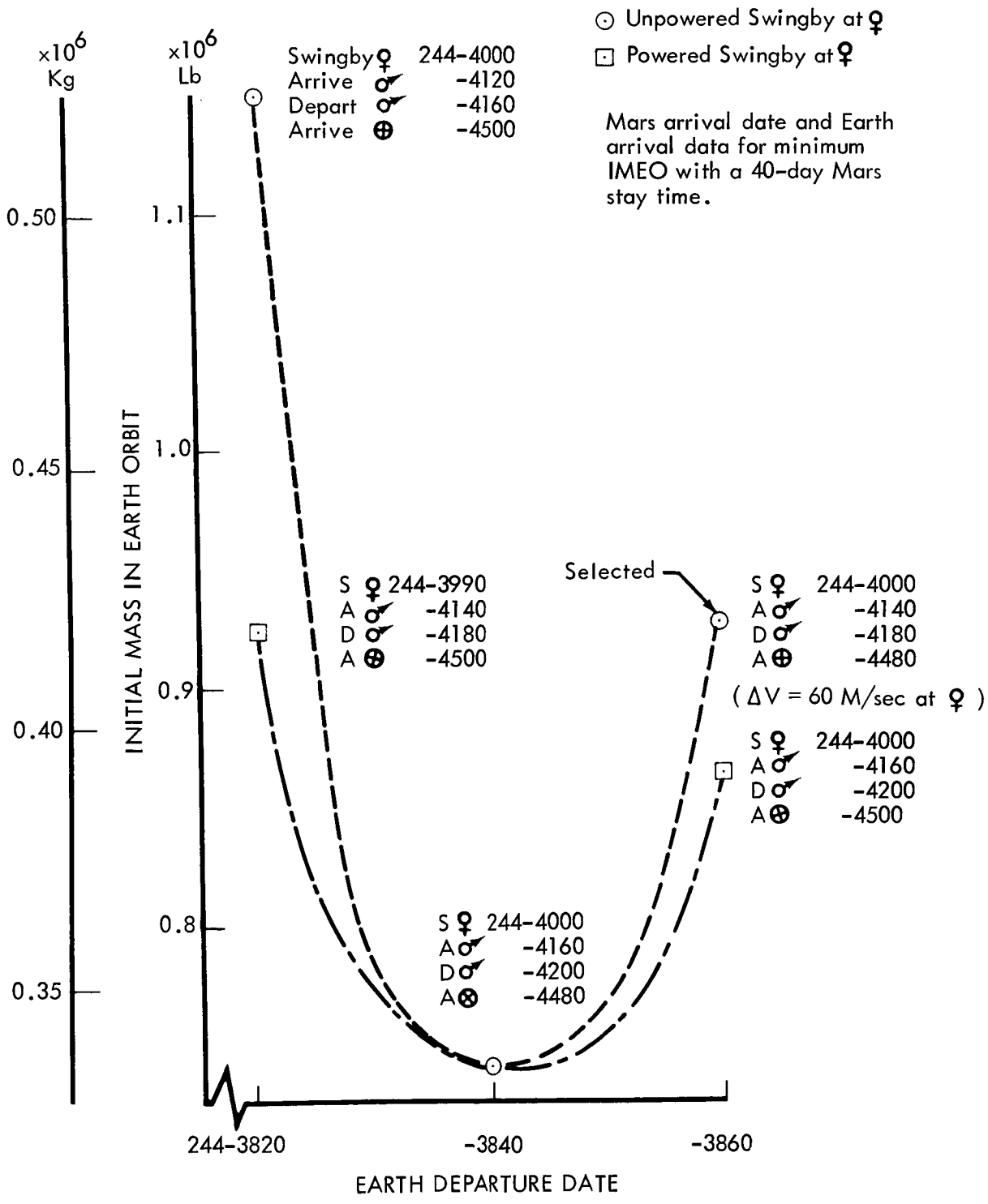


Figure 2.1-24: 1980 MARS TRAJECTORIES WITH VENUS SWINGBY

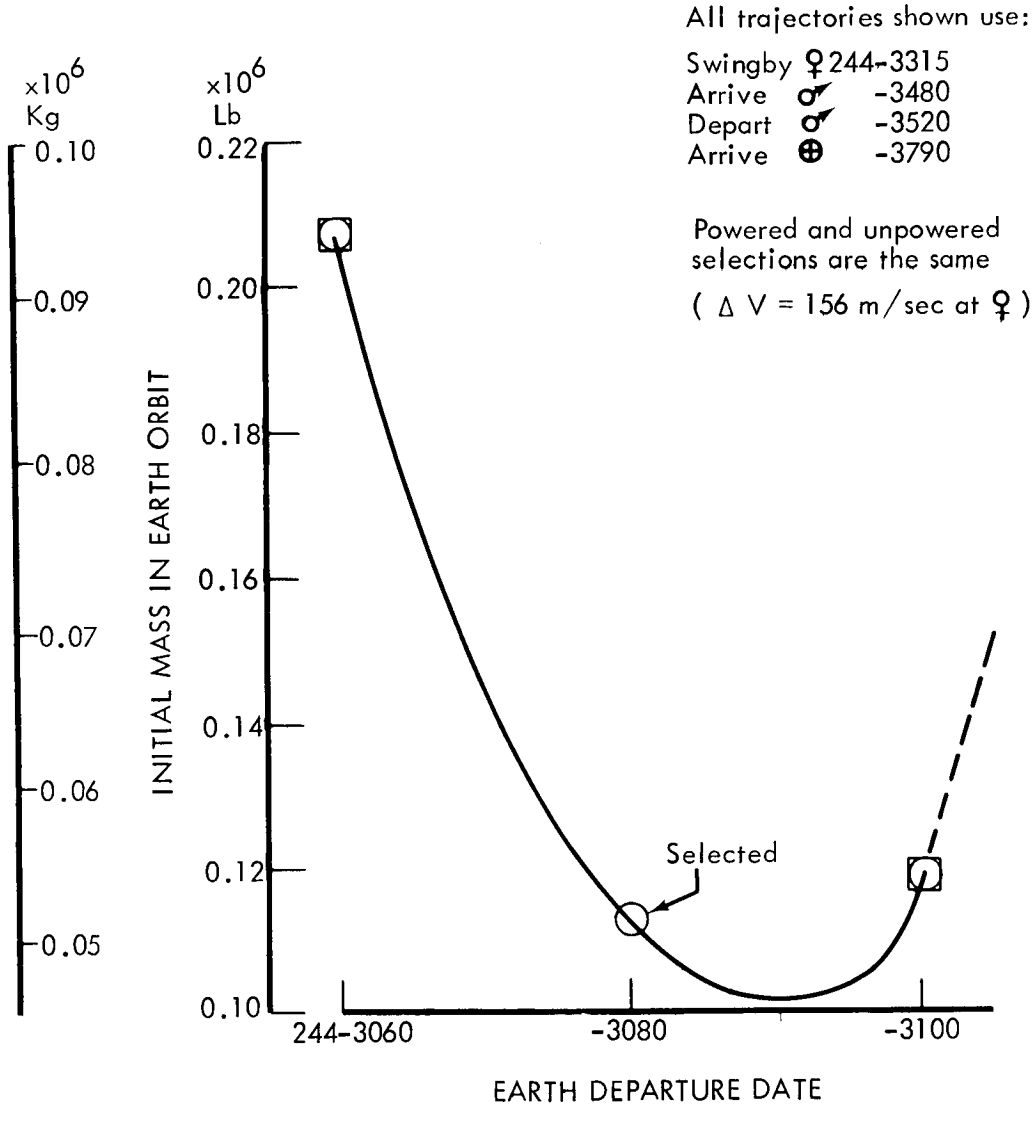


Figure 2.1-25: 1978 MARS TRAJECTORIES WITH VENUS SWINGBY

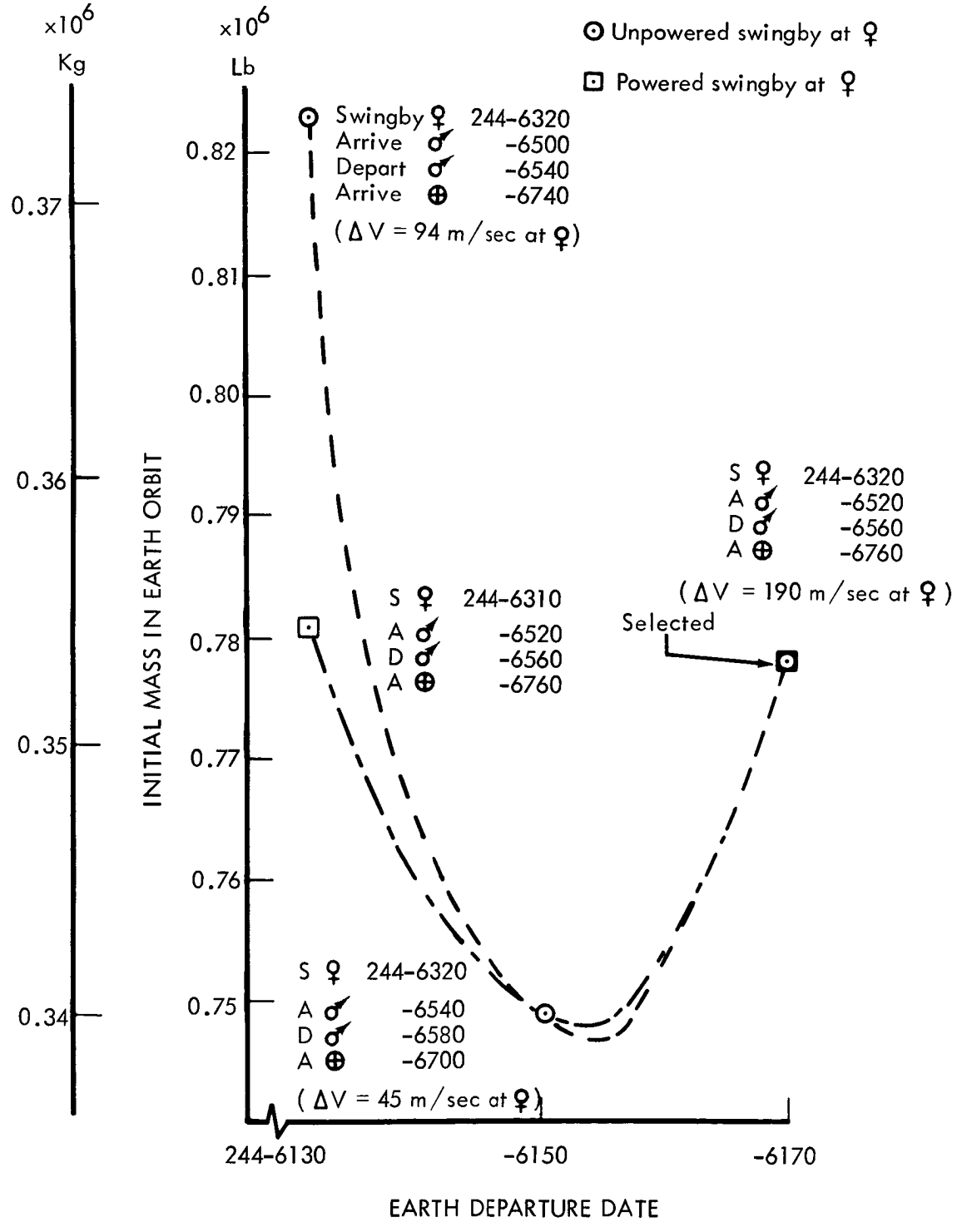


Figure 2.1-26: 1986 MARS TRAJECTORIES WITH VENUS SWINGBY

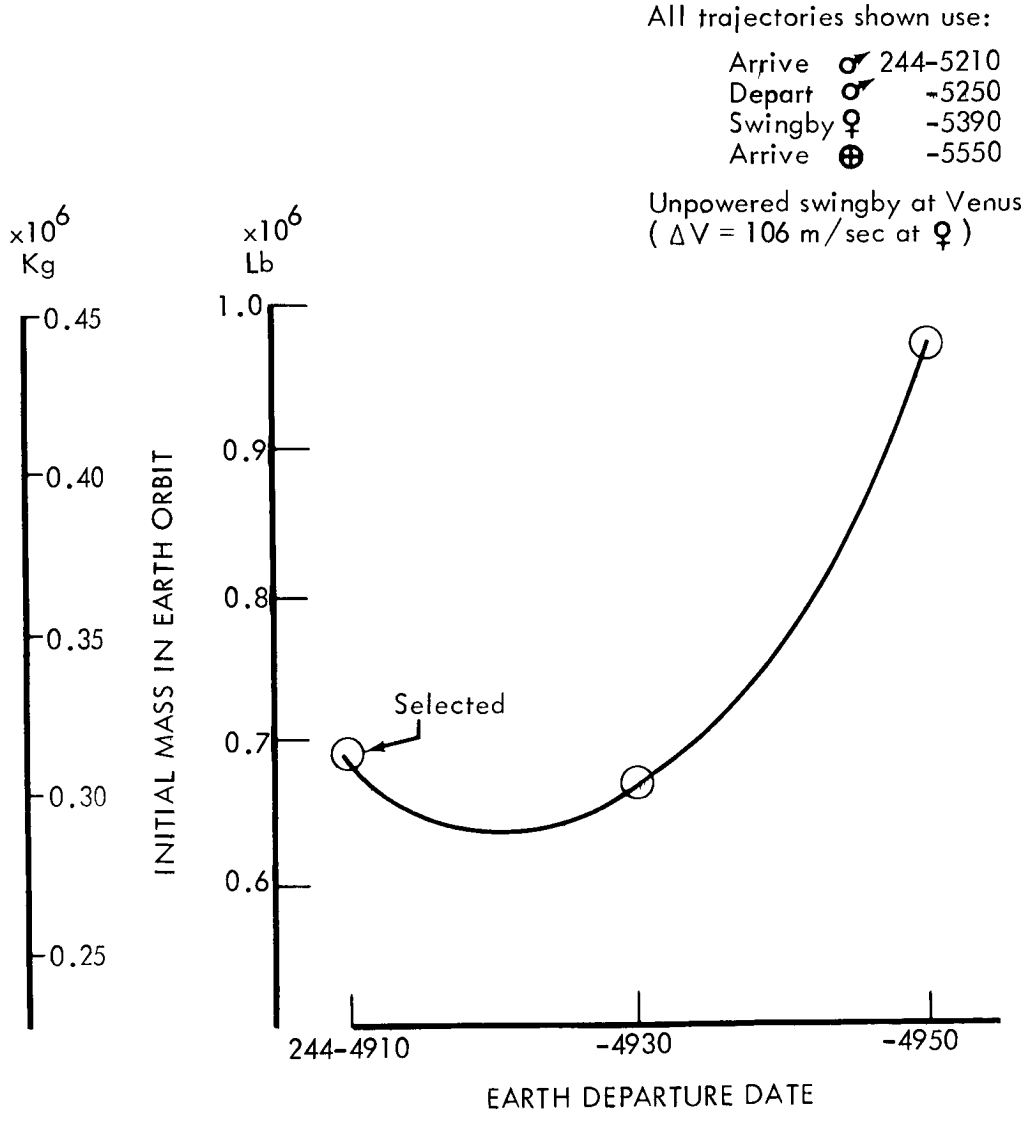


Figure 2.1-27: 1982 MARS TRAJECTORIES WITH VENUS SWINGBY

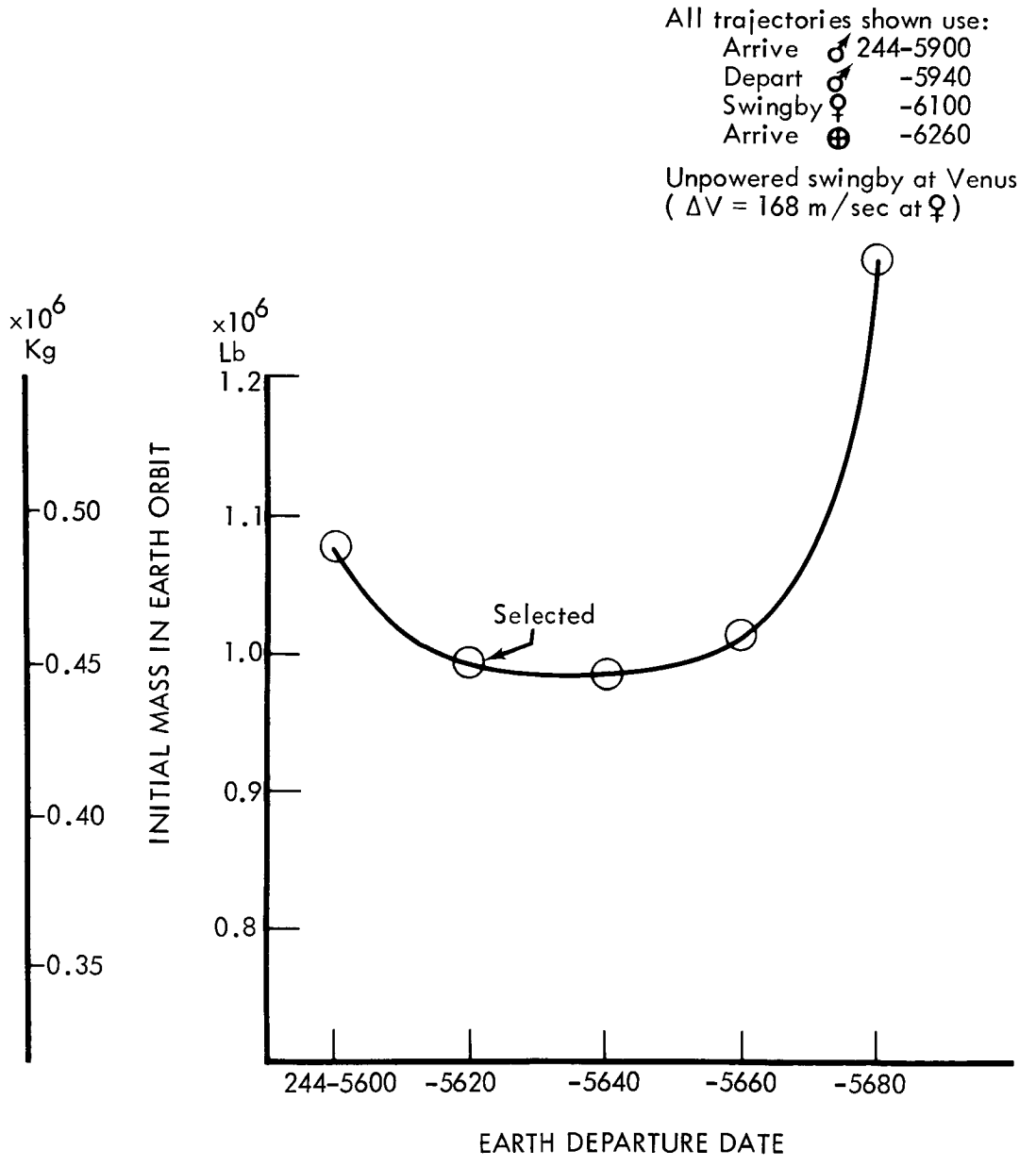


Figure 2.1-28: 1984 MARS TRAJECTORIES WITH VENUS SWINGBY

APPENDIX A-4

1986 MARS OPPOSITION MISSION CHARTS--Aerobraker and Propulsion Trajectories. Figures 2.2-21 through 2.2-31. Explanation of these curves is given in Section 2.2.3.

Note:

The following notation is used to identify mission trip times.

Outbound triptime (days) = XXX

$$\frac{\text{Total mission triptime (days)}}{\text{Outbound triptime (days)}} = \frac{XXX}{XXX}$$

Planet staytime (days) = 40

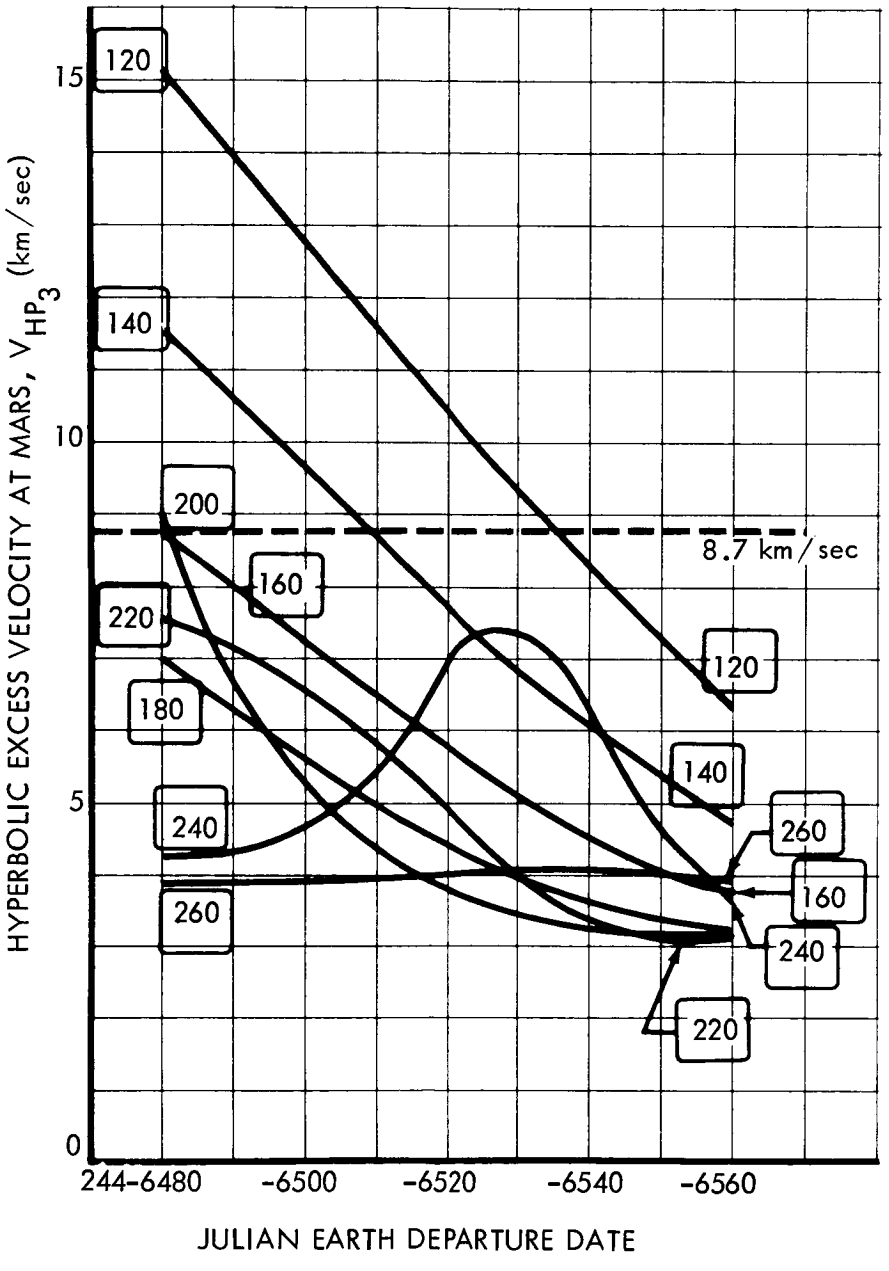


Figure 2.2-21: 1986 MARS OPPOSITION

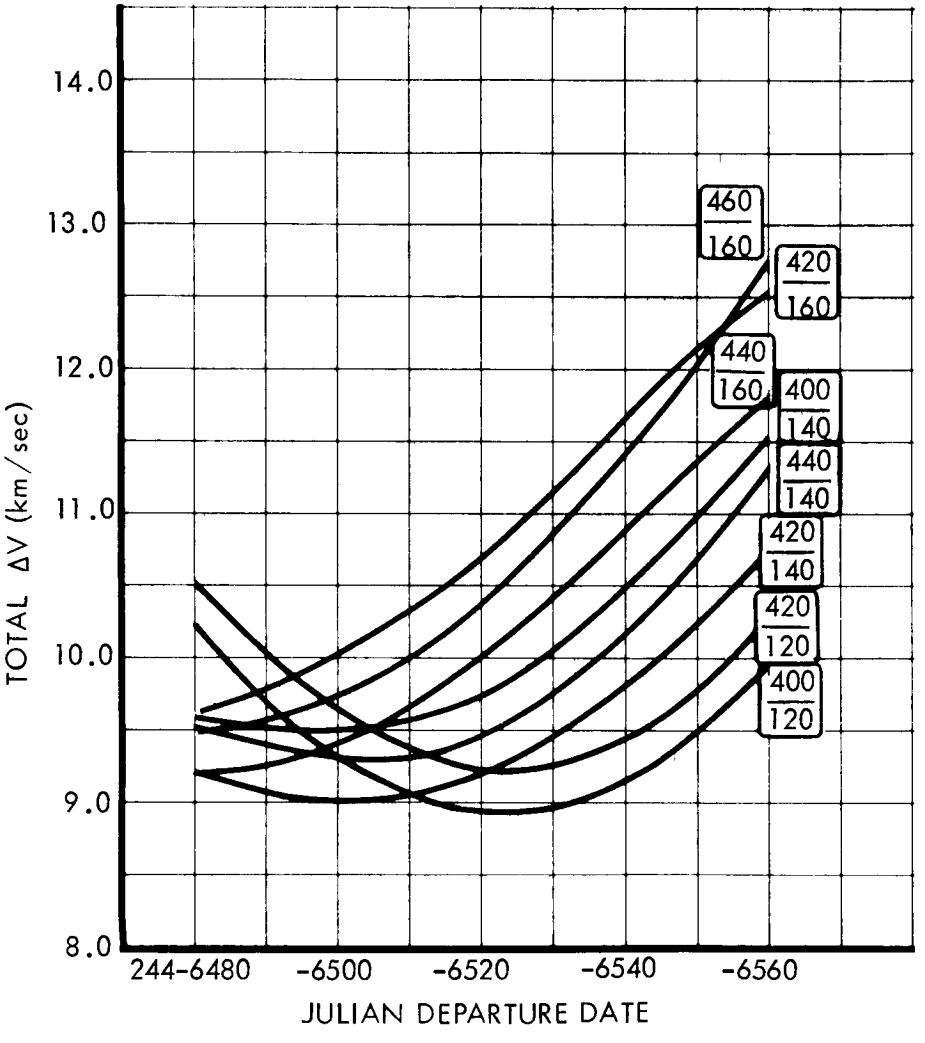


Figure 2.2-22: 1986 MARS OPPOSITION — Aerobraker

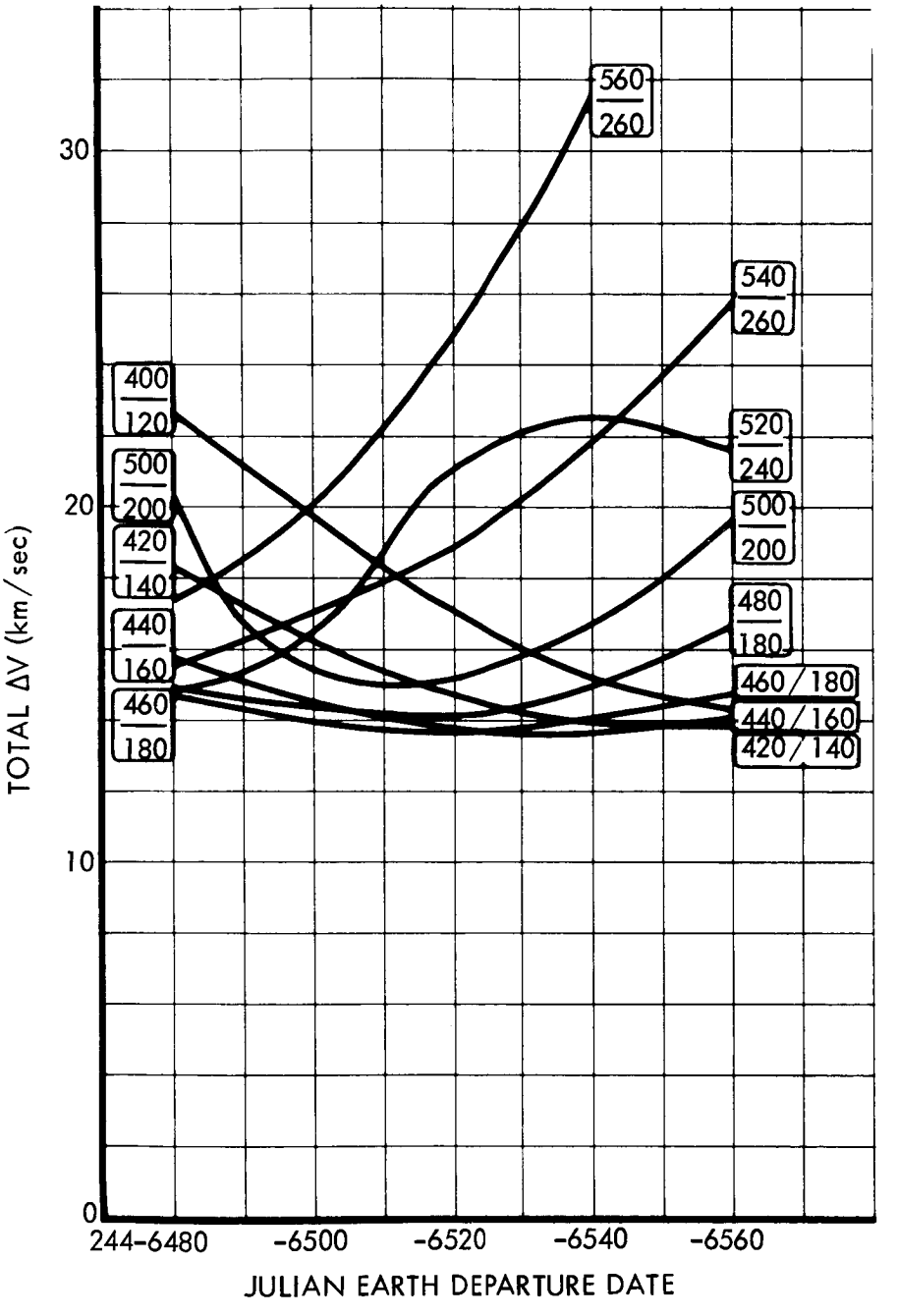


Figure 2.2-23: 1986 MARS OPPOSITION — All-Propulsive

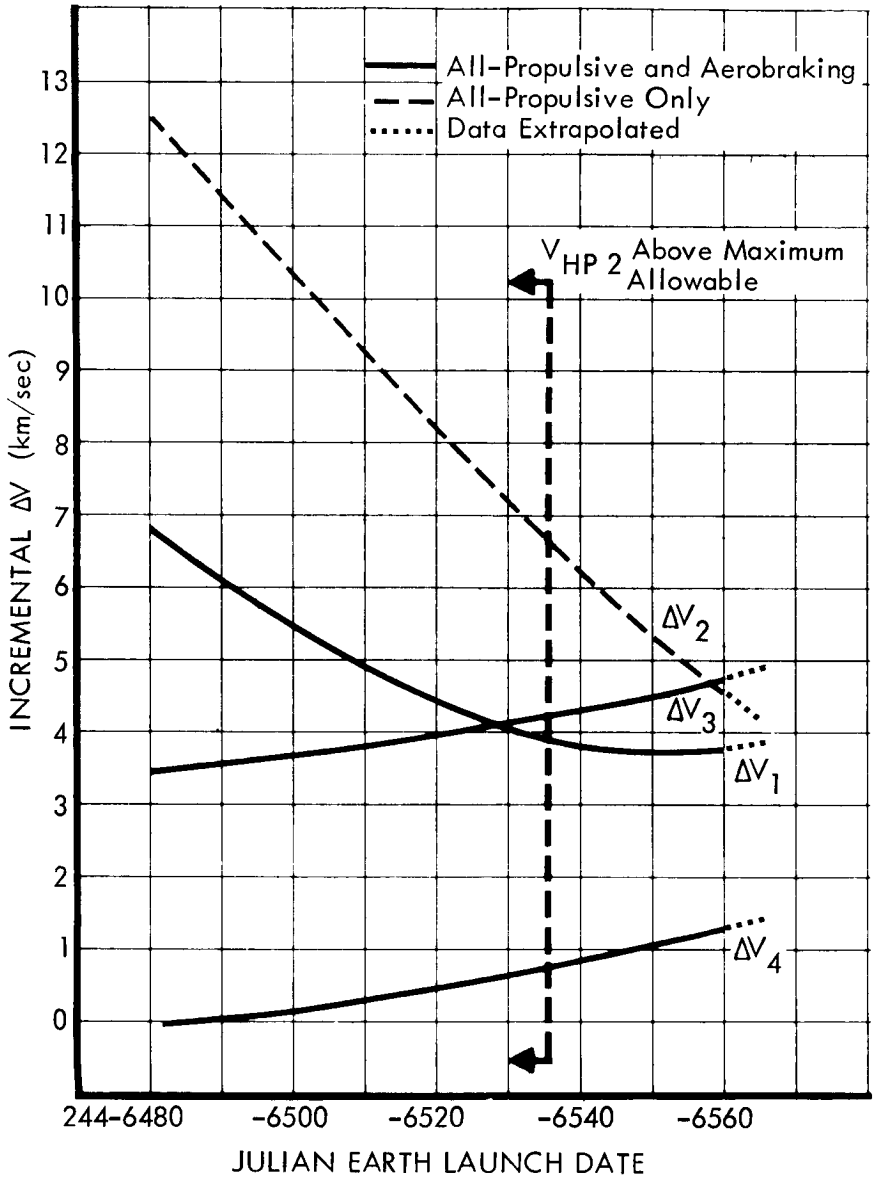


Figure 2.2-24: 1986 MARS OPPOSITION — 400/120 Mission

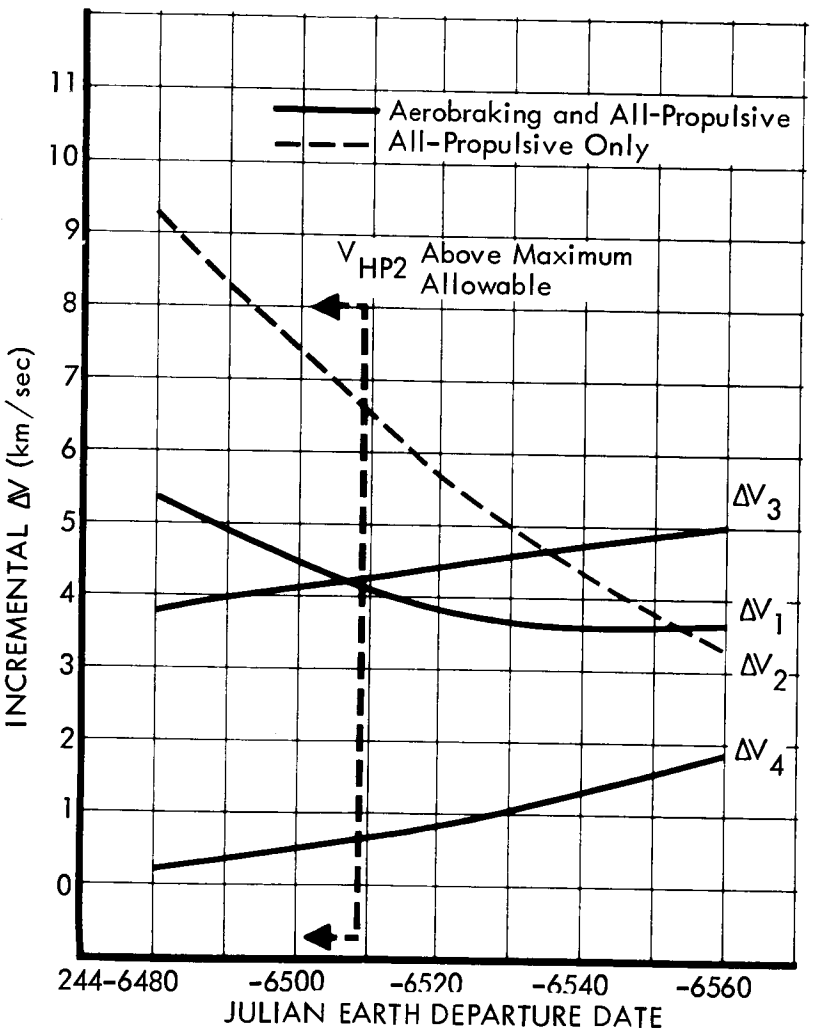


Figure 2.2-25: 1986 MARS OPPOSITION — 420/140 Mission

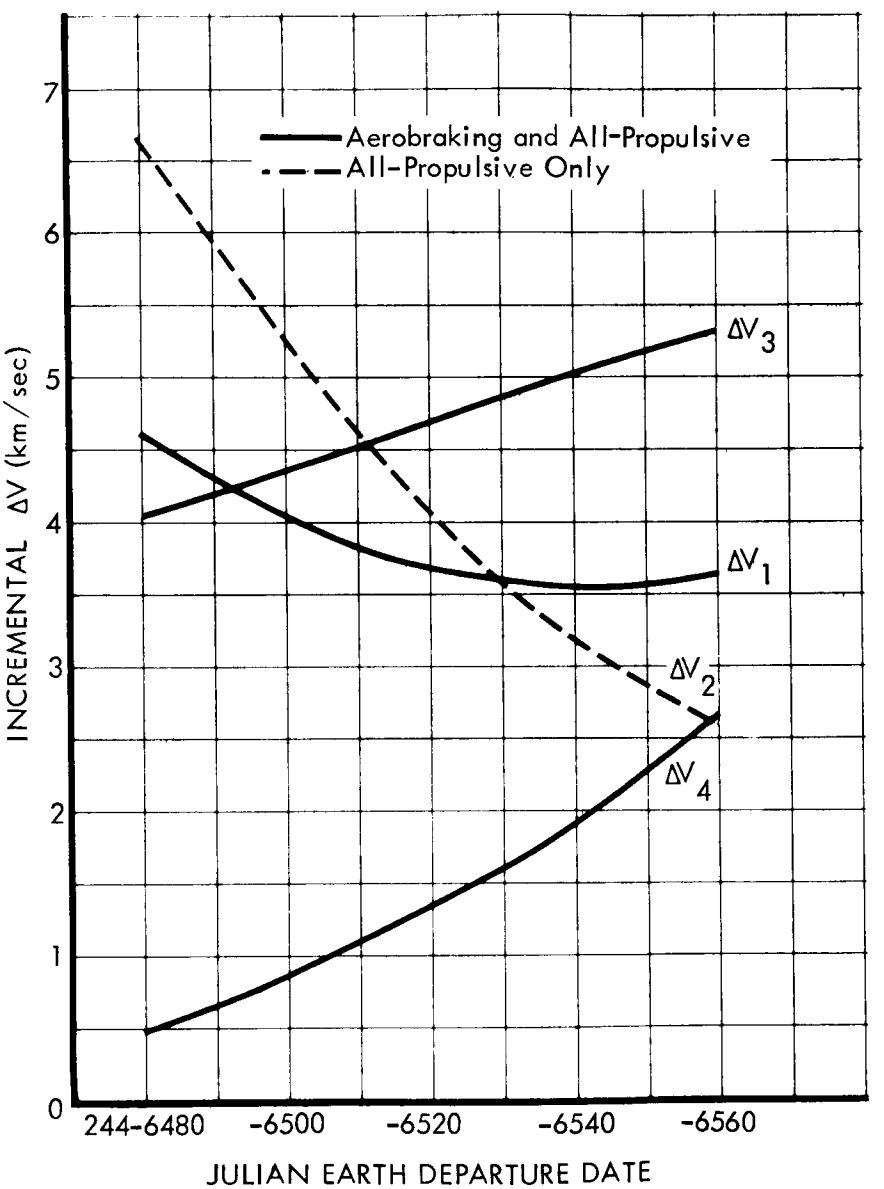


Figure 2.2-26: 1986 MARS OPPOSITION — 440/160 Mission

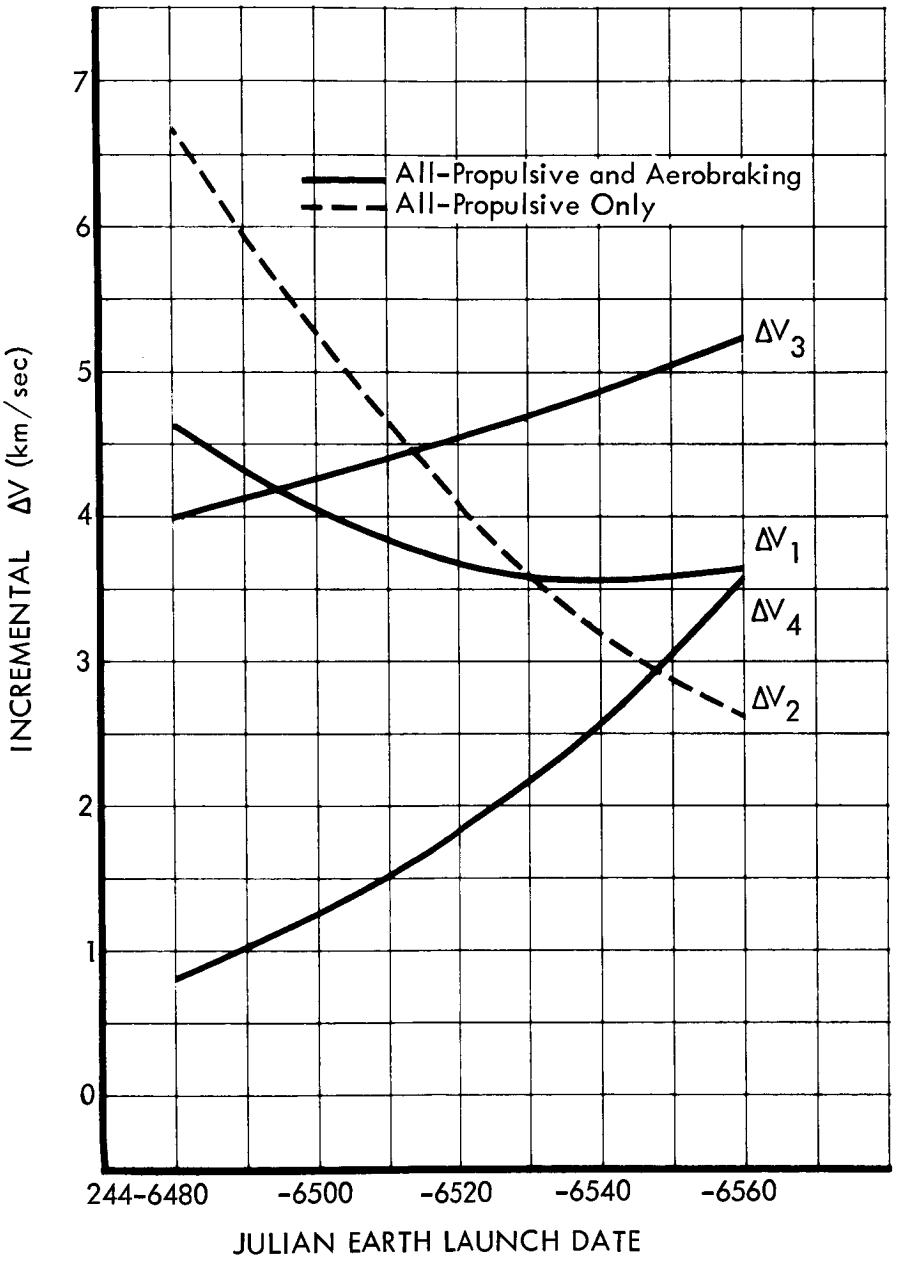


Figure 2.2-27: 1986 MARS OPPOSITION — 460/160 Mission

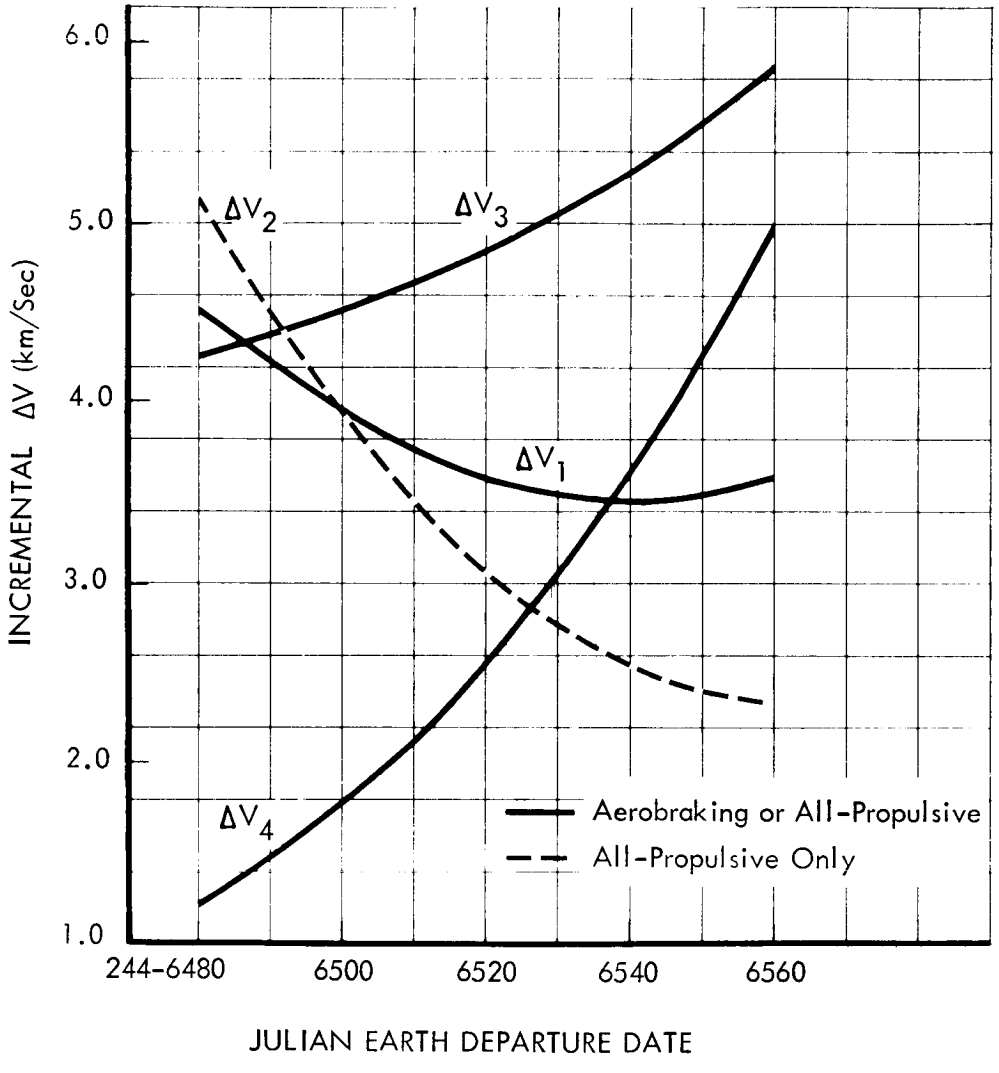


Figure 2.2-28: 1986 MARS OPPOSITION — 480/180 Mission

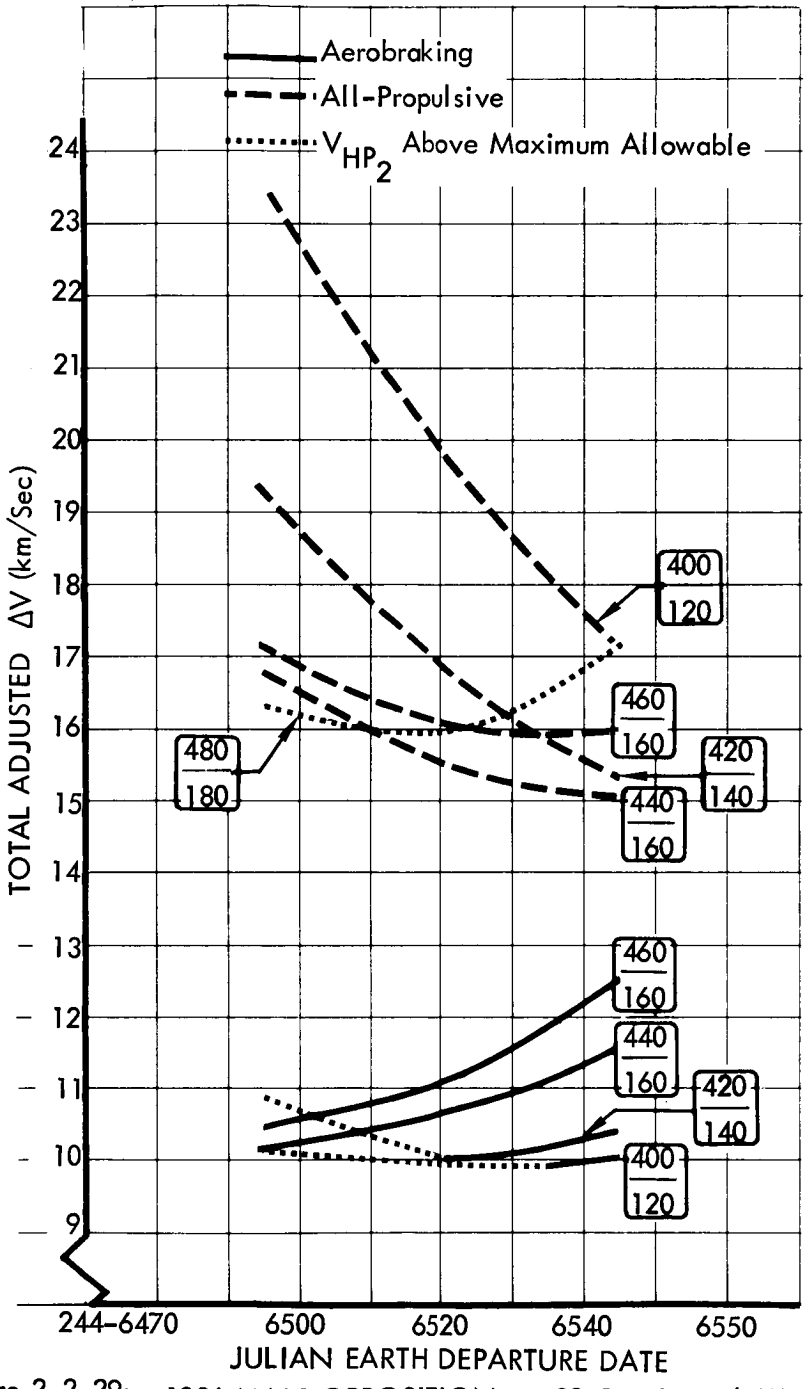


Figure 2.2-29: 1986 MARS OPPOSITION — 30-Day Launch Window

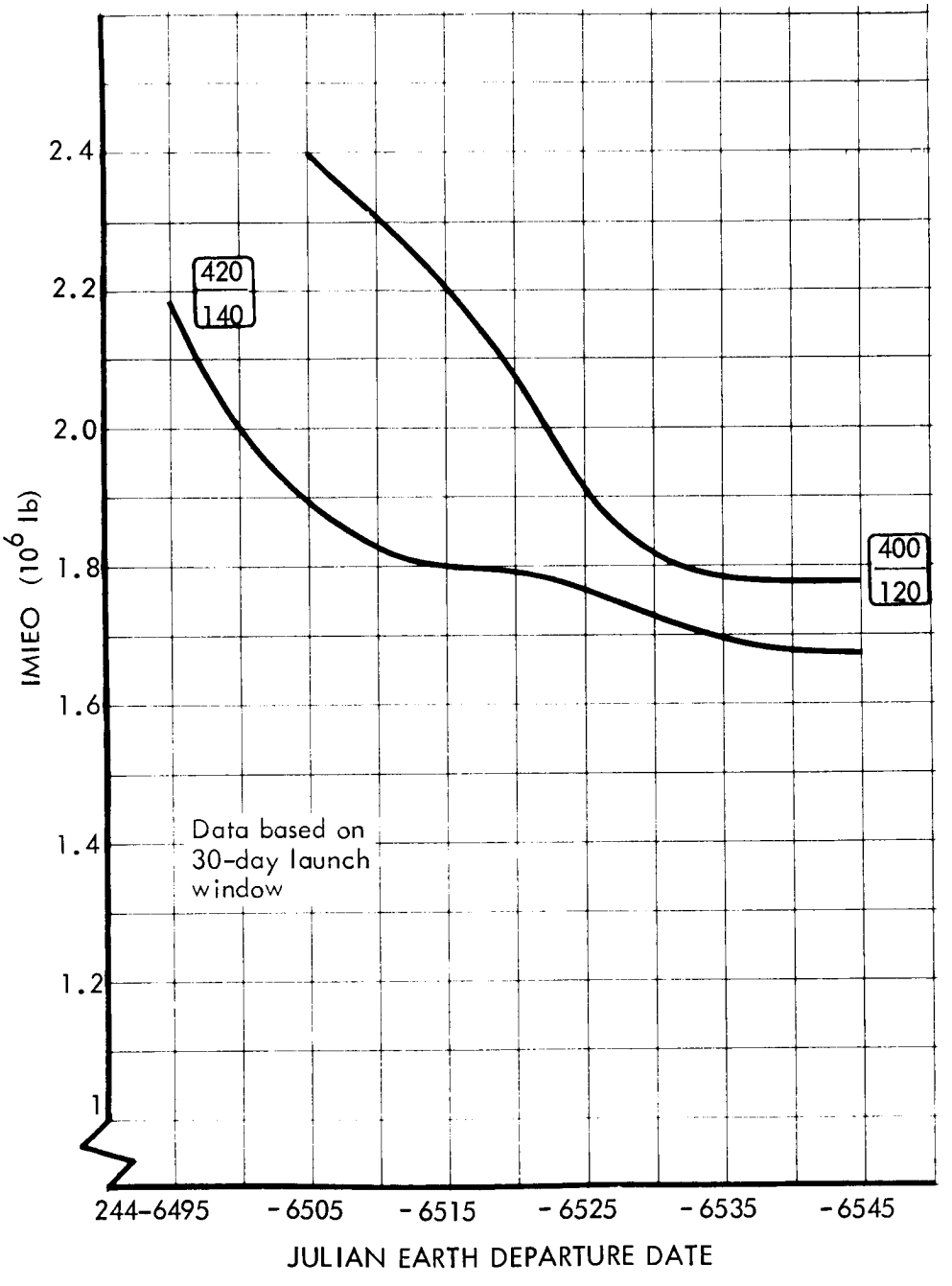


Figure 2.2-30: 1986 MARS OPPOSITION — NAC

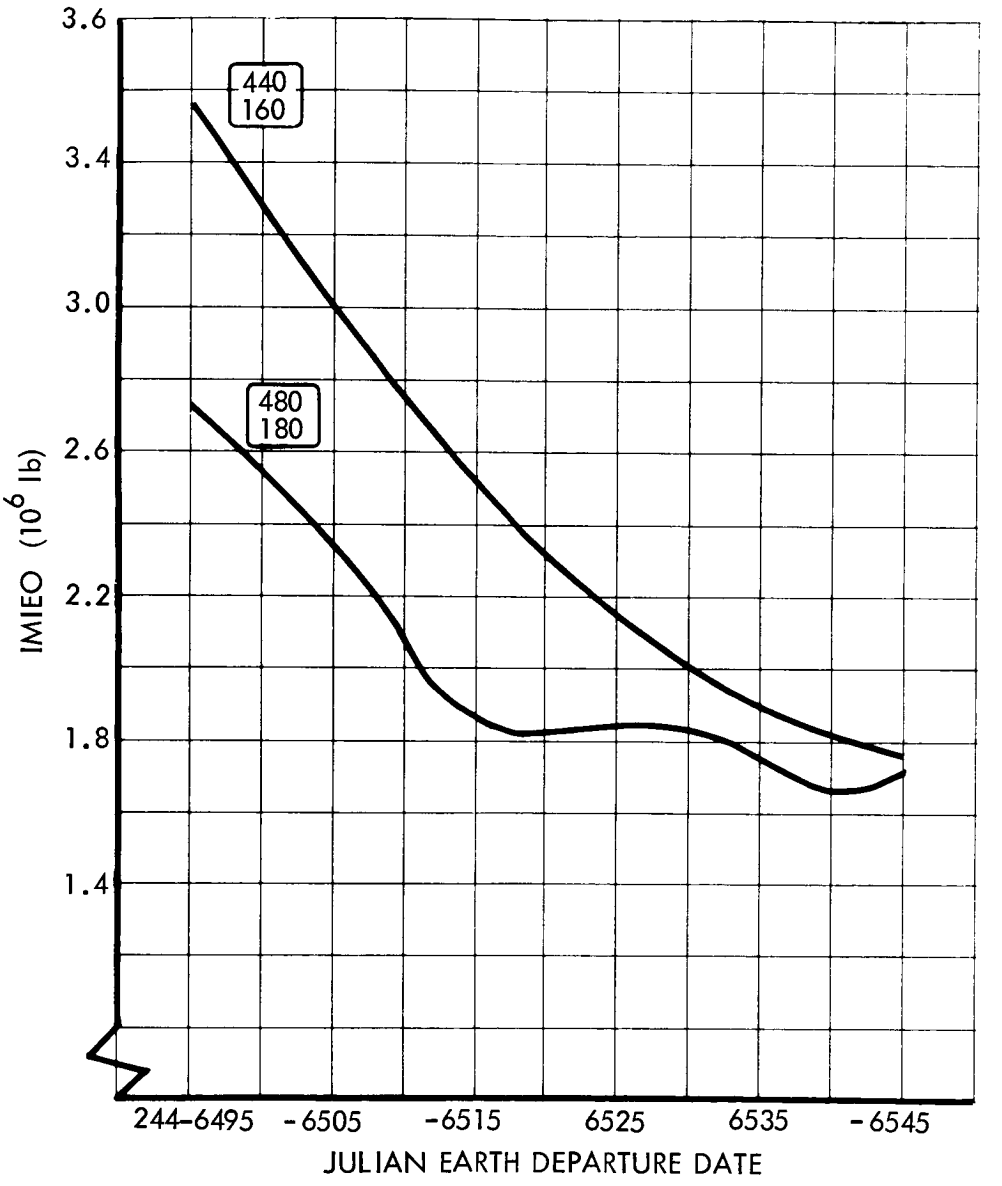


Figure 2.2-31: 1986 MARS OPPOSITION — NNN

APPENDIX A-5

1986 MARS CONJUNCTION MISSION CHARTS--Aerobraker and All-Propulsive Trajectories. Figures 2.2-52 through 2.2-40. Explanation of these curves is given in Section 2.2.3.

Note:

The following notation is used to identify mission triptimes.

Outbound triptime (days) = XXX

Outbound triptime/planet staytime/inbound triptime = XXX/XXX/XXX

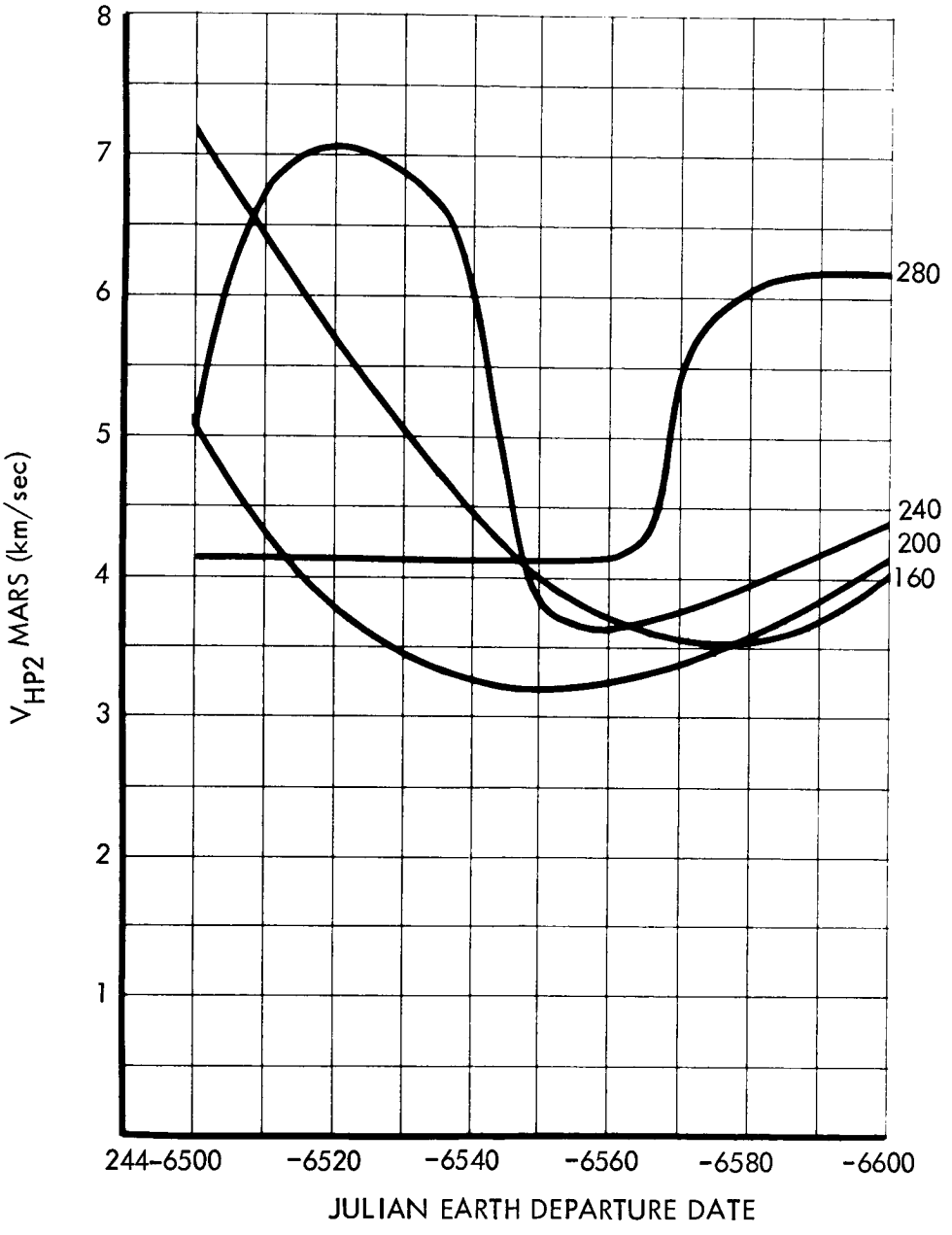


Figure 2.2-32: 1986 MARS CONJUNCTION

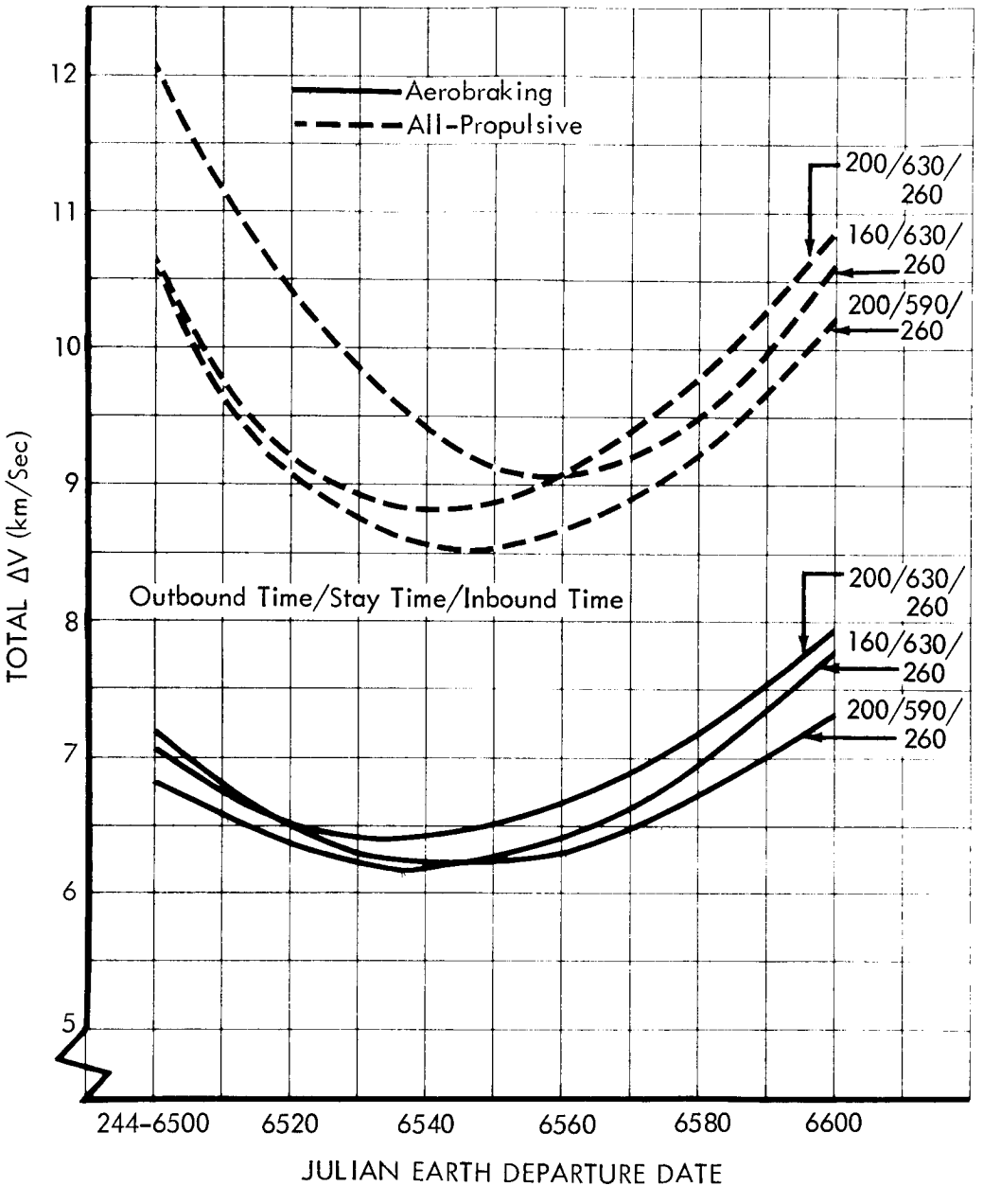


Figure 2.2-33: 1986 MARS CONJUNCTION

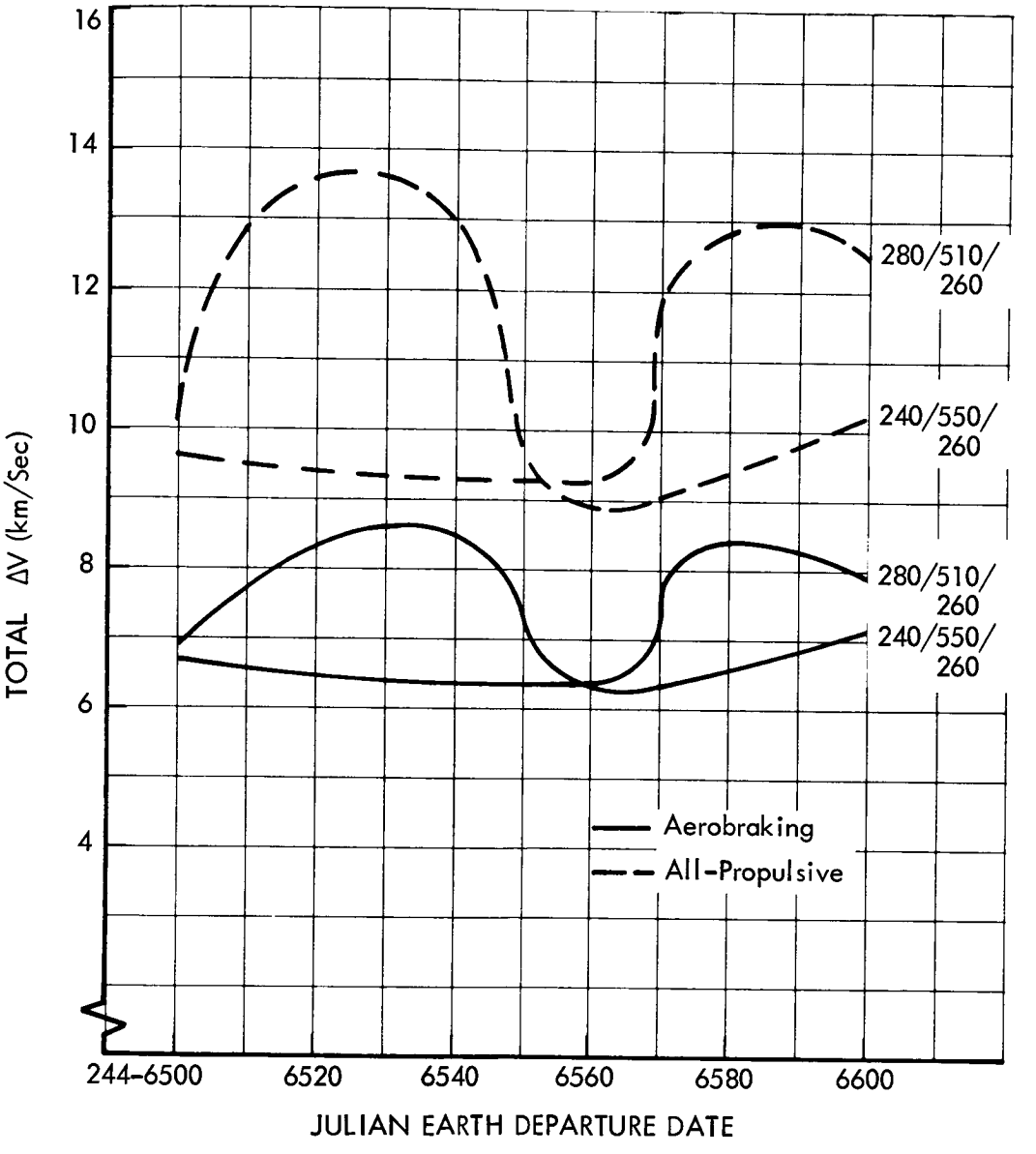


Figure 2.2-34: 1986 MARS CONJUNCTION

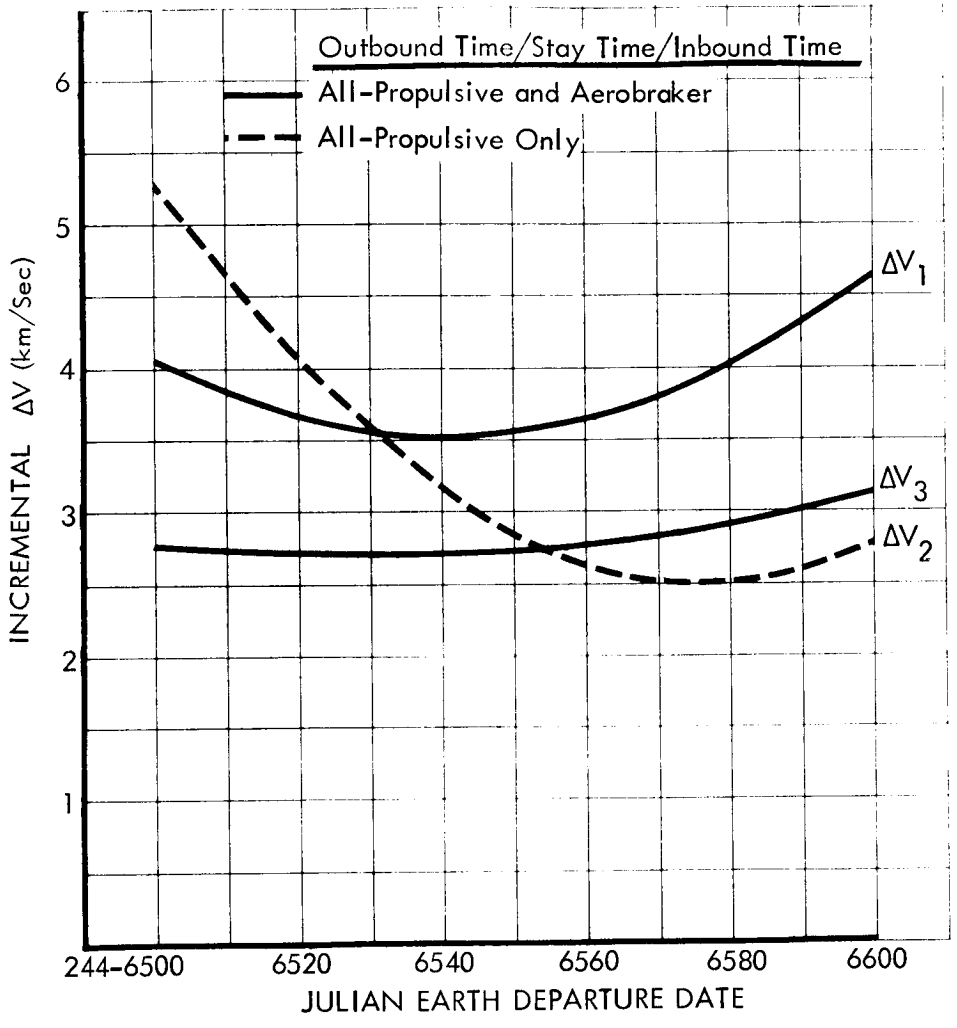


Figure 2.2-35: 1986 MARS CONJUNCTION — 160/630/260 Mission

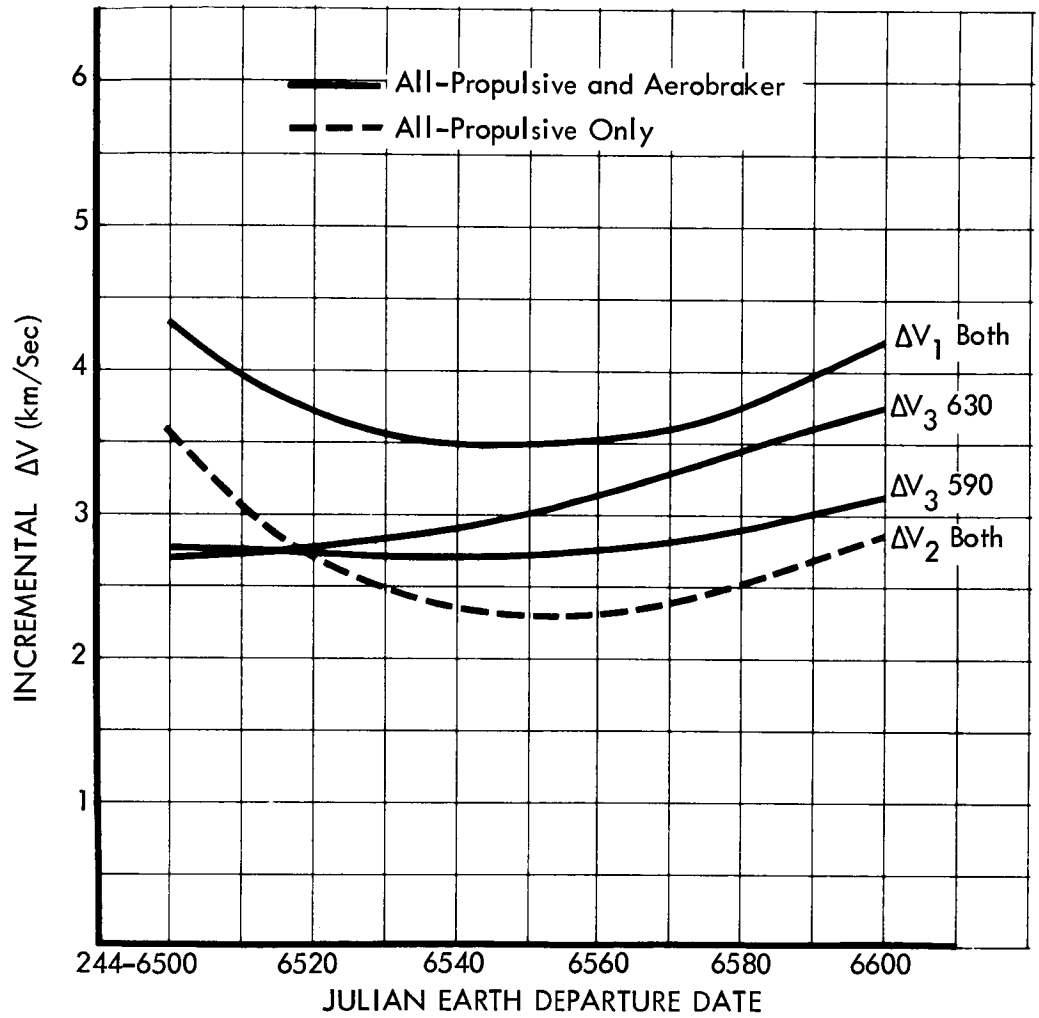


Figure 2.2-36: 1986 MARS CONJUNCTION
— 200/590/260 & 200/630/260 Missions

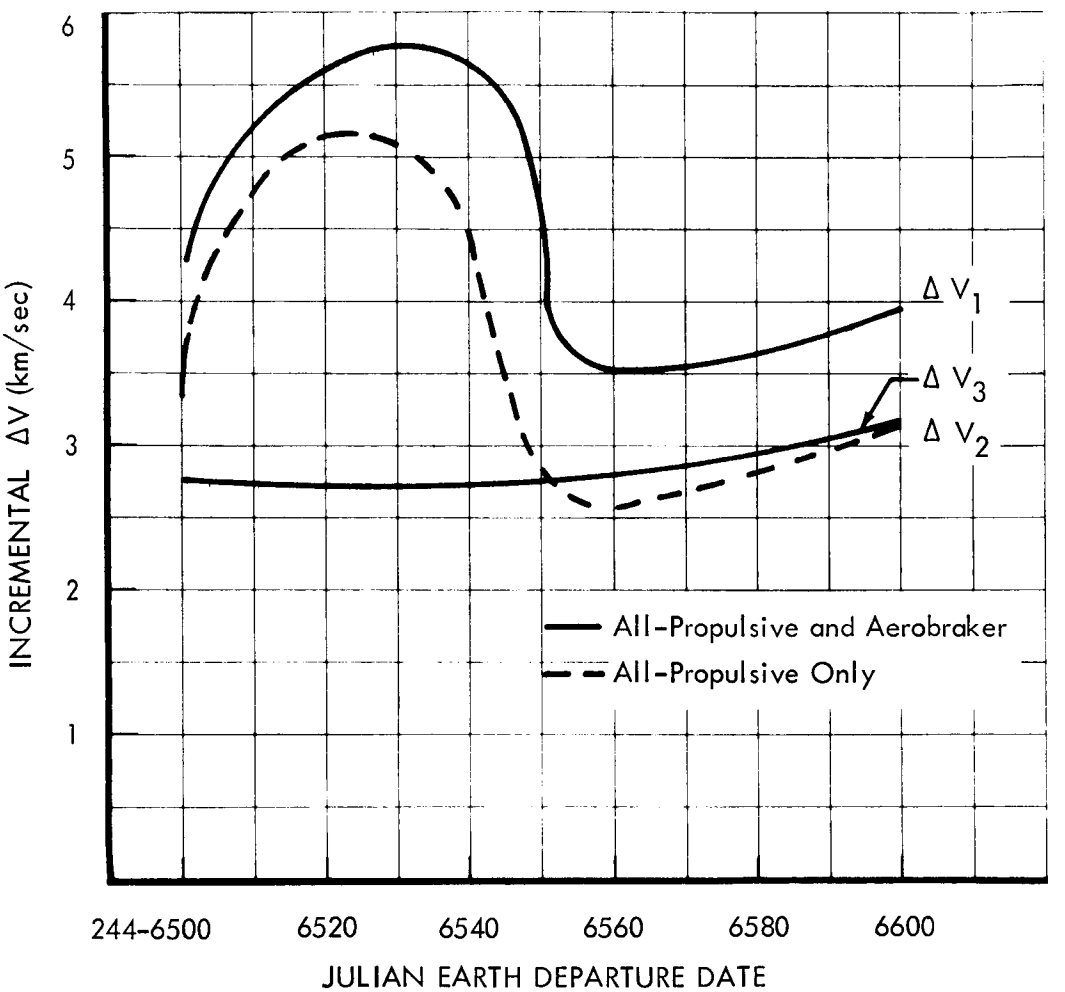


Figure 2.2-37: 1986 MARS CONJUNCTION — 240/550/260 Mission

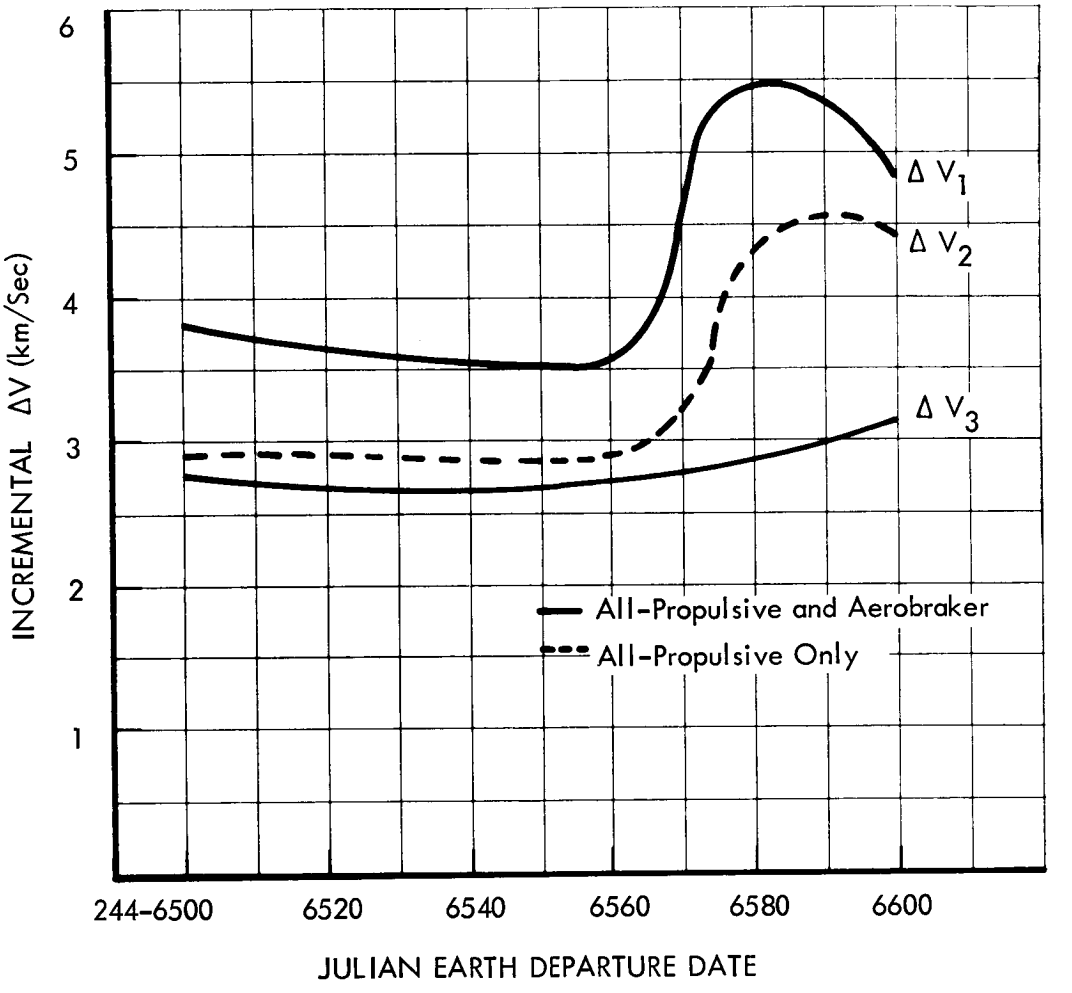


Figure 2.2-38: 1986 MARS CONJUNCTION — 280/510/260 Mission

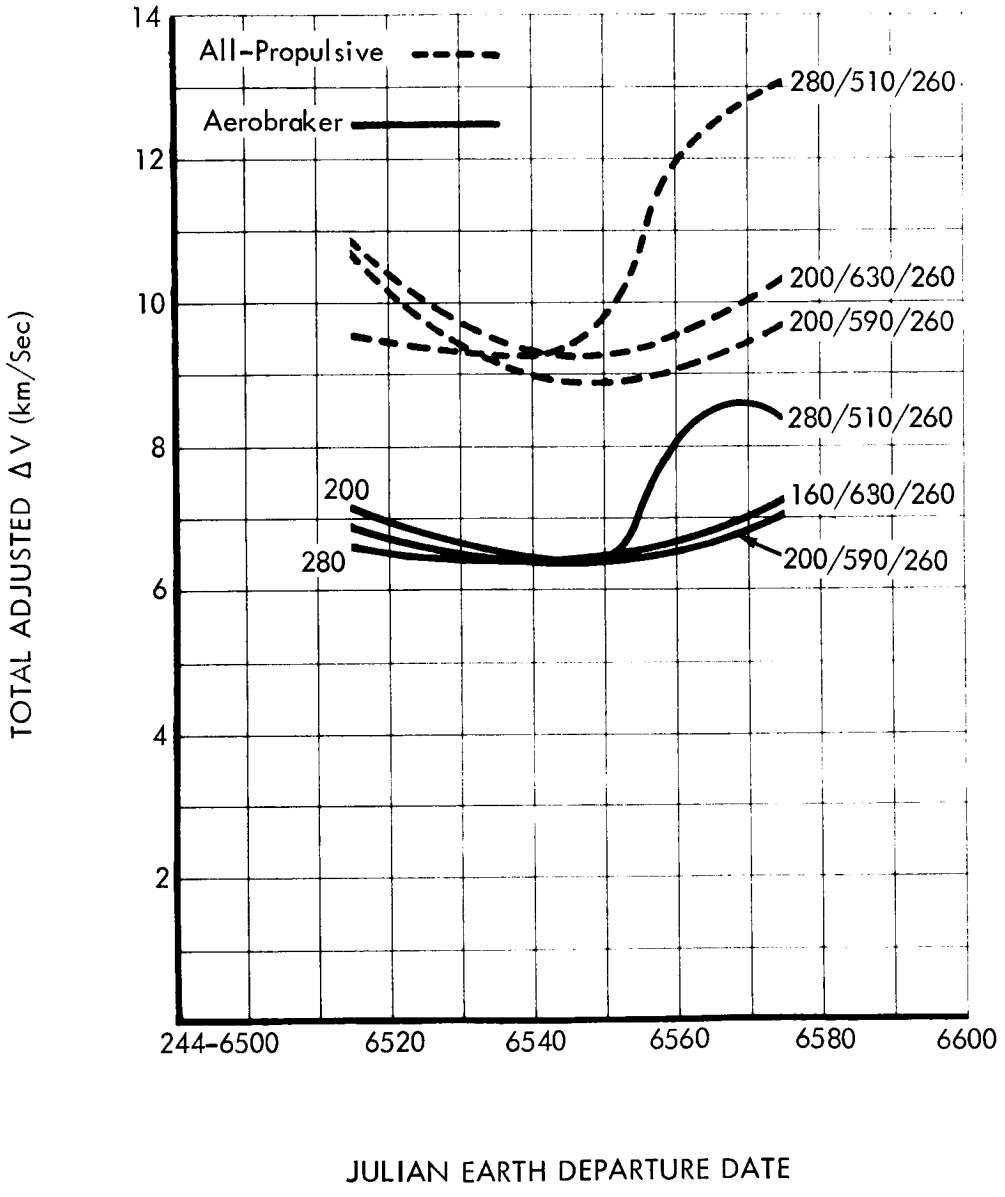


Figure 2.2-39: 1986 MARS CONJUNCTION

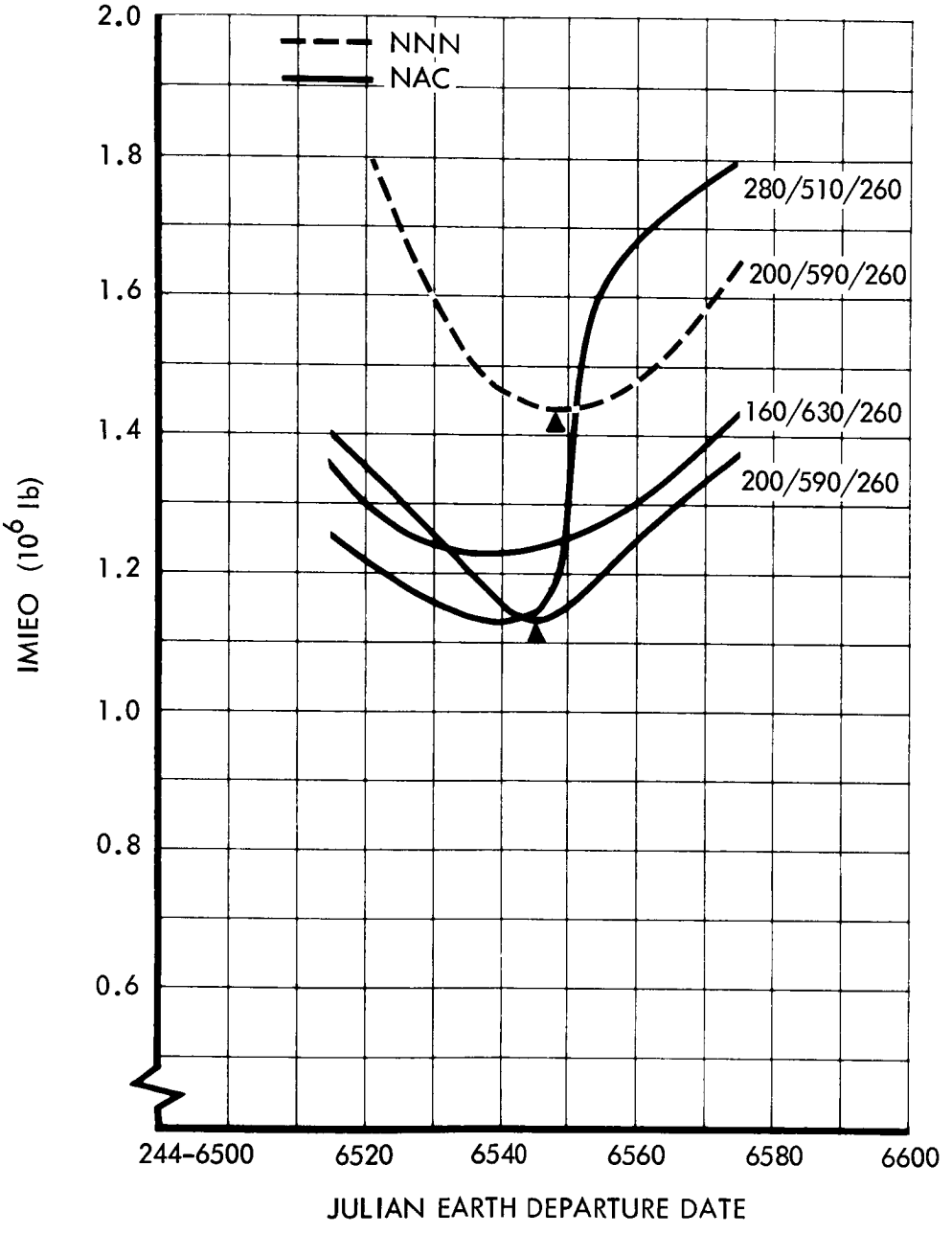


Figure 2.2-40: 1986 MARS CONJUNCTION

APPENDIX A-6

1982 MARS WITH VENUS SWINGBY MISSION CHARTS.
Figure 2.2-41 through 2.2-47. Explanation of these curves is given in Section 2.2.3.

Note:

The following notation is used to identify mission triptimes and dates.

Triptime to Venus/triptime from Venus to Earth = XXX/XXXXX

Triptime to Venus/triptime from Venus to Earth (departure date) = XXX/XXX (XXXX)

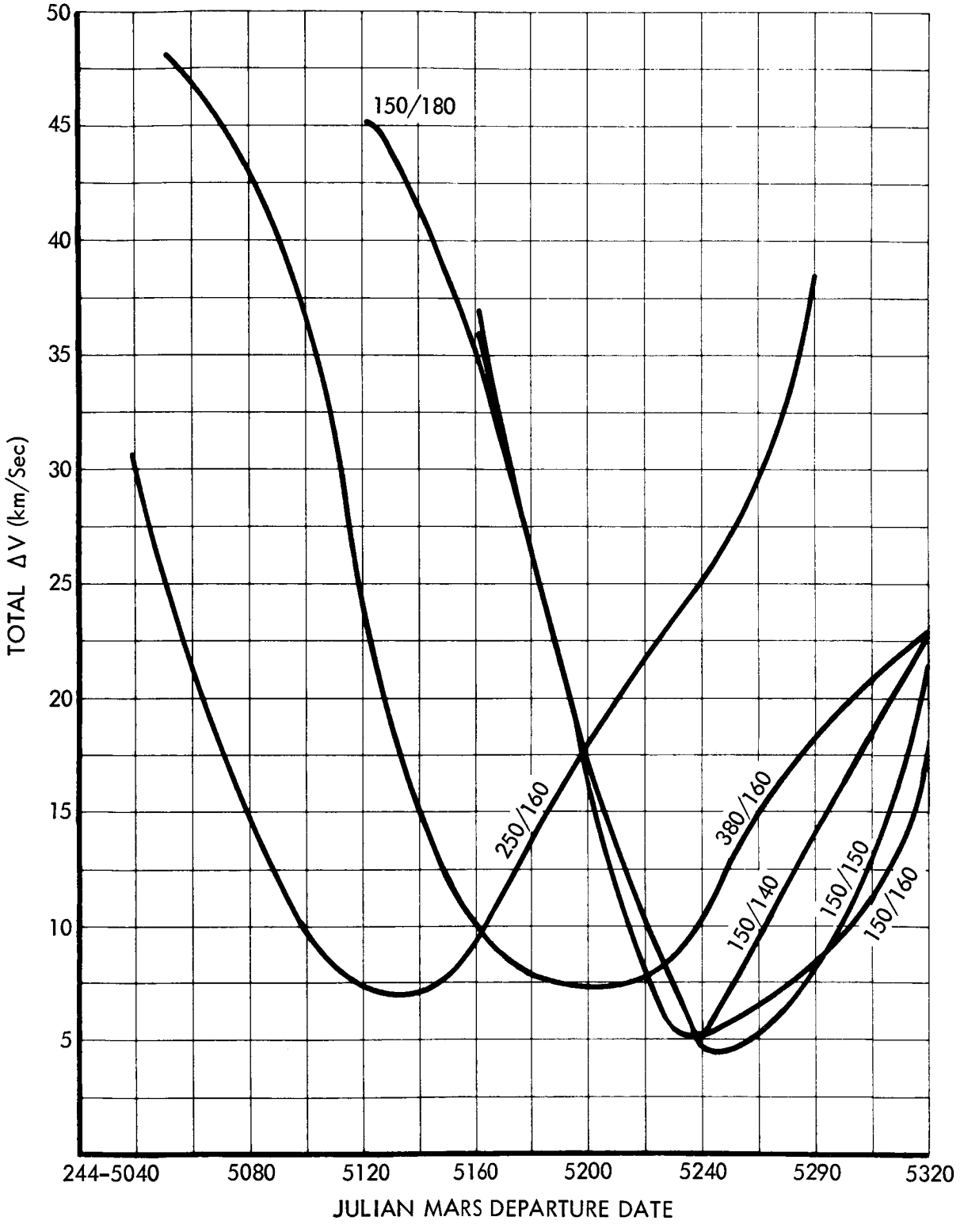


Figure 2.2-41: 1982 SWINGBY, INBOUND

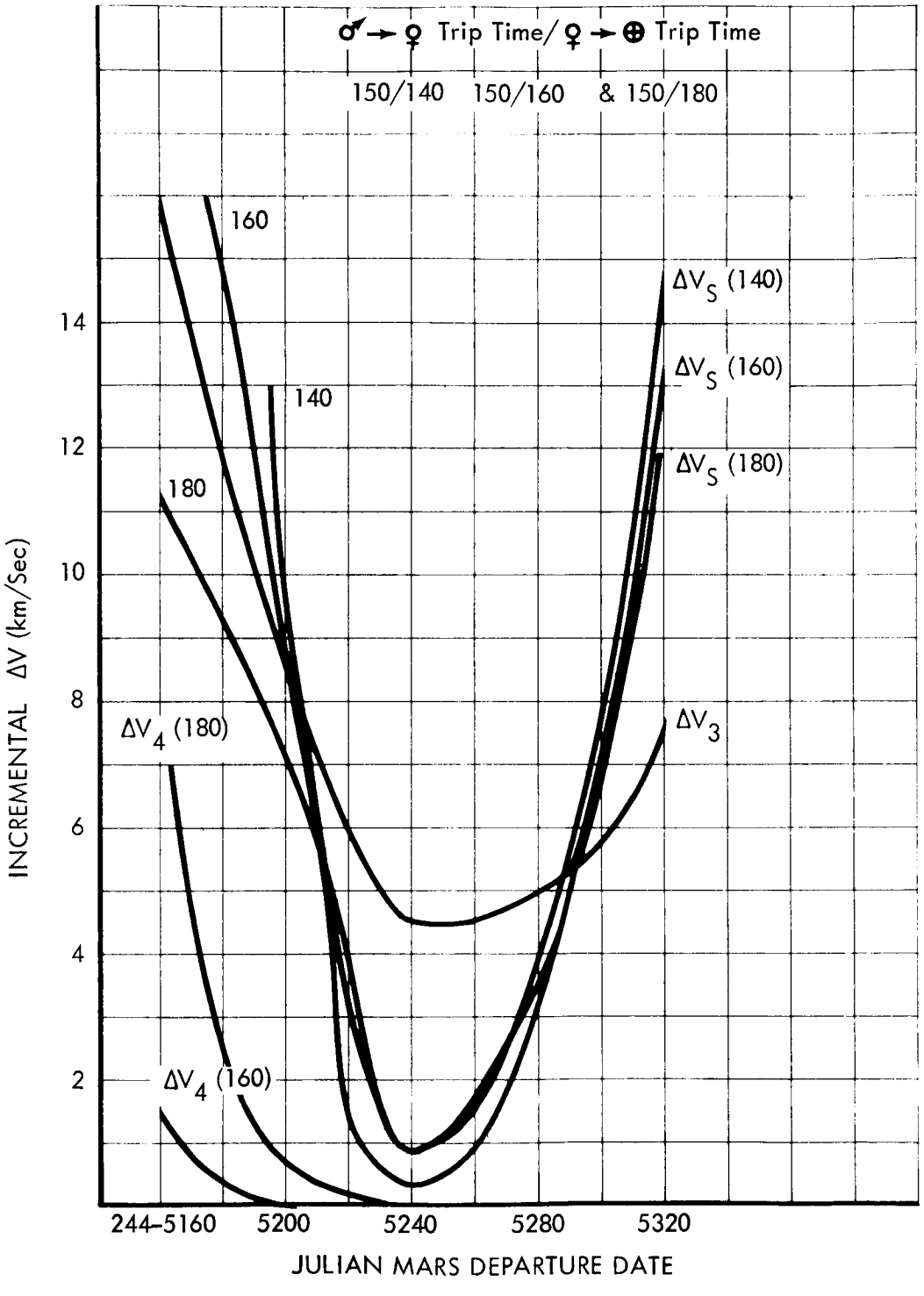


Figure 2.2-42: 1982 SWINGBY, INBOUND

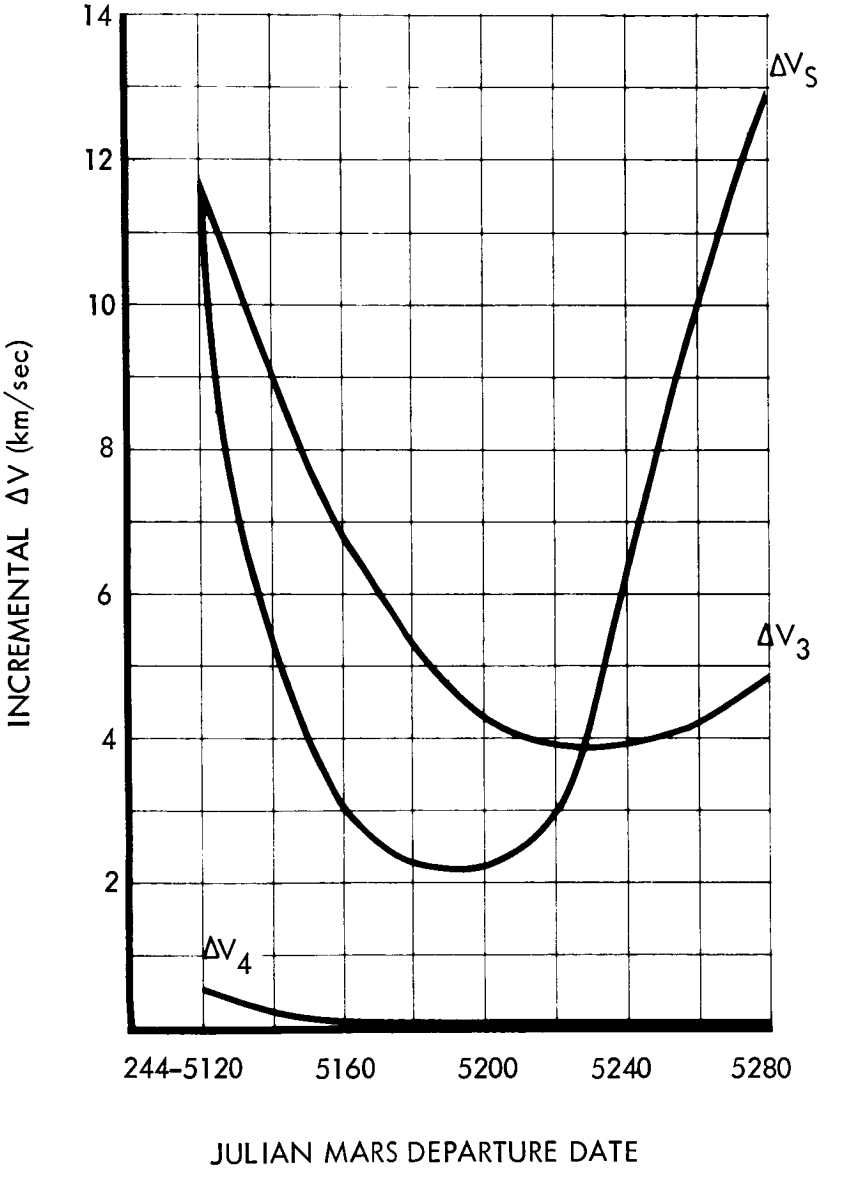


Figure 2.2-43: 1982 SWINGBY — 200/160, INBOUND

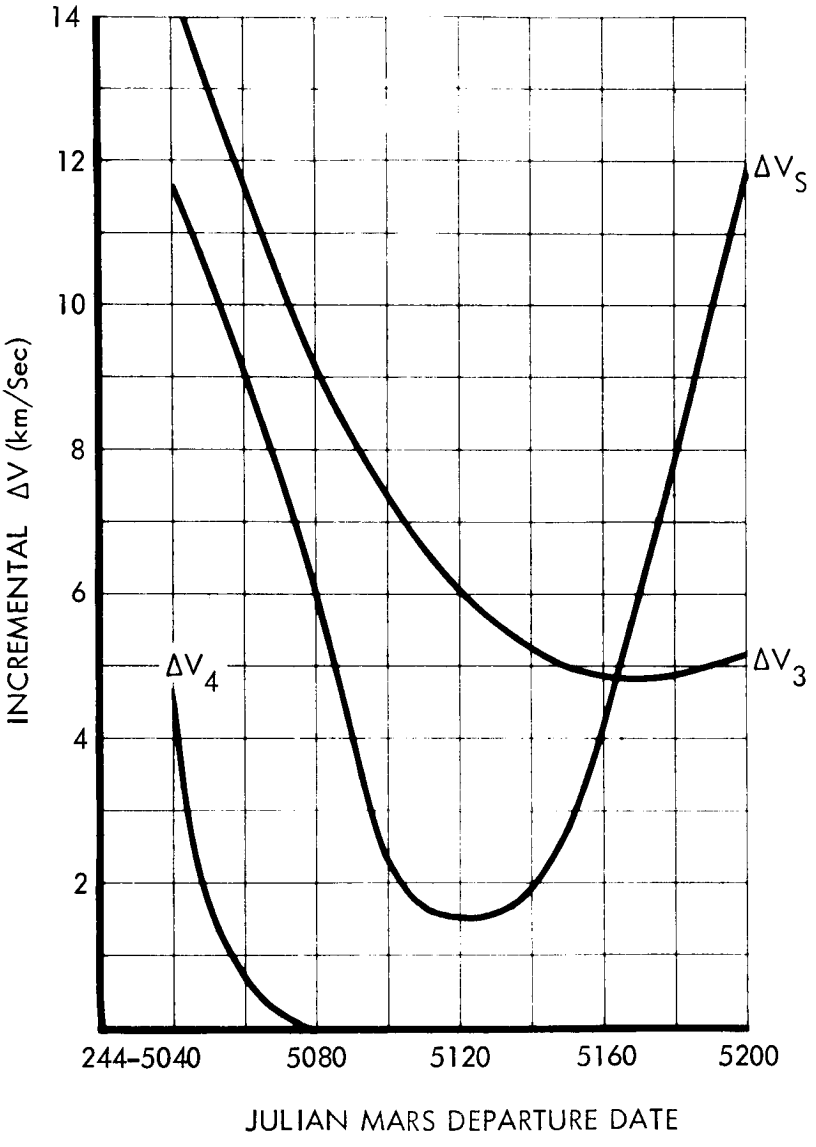


Figure 2.2-44: 1982 SWINGBY — 250/160 INBOUND

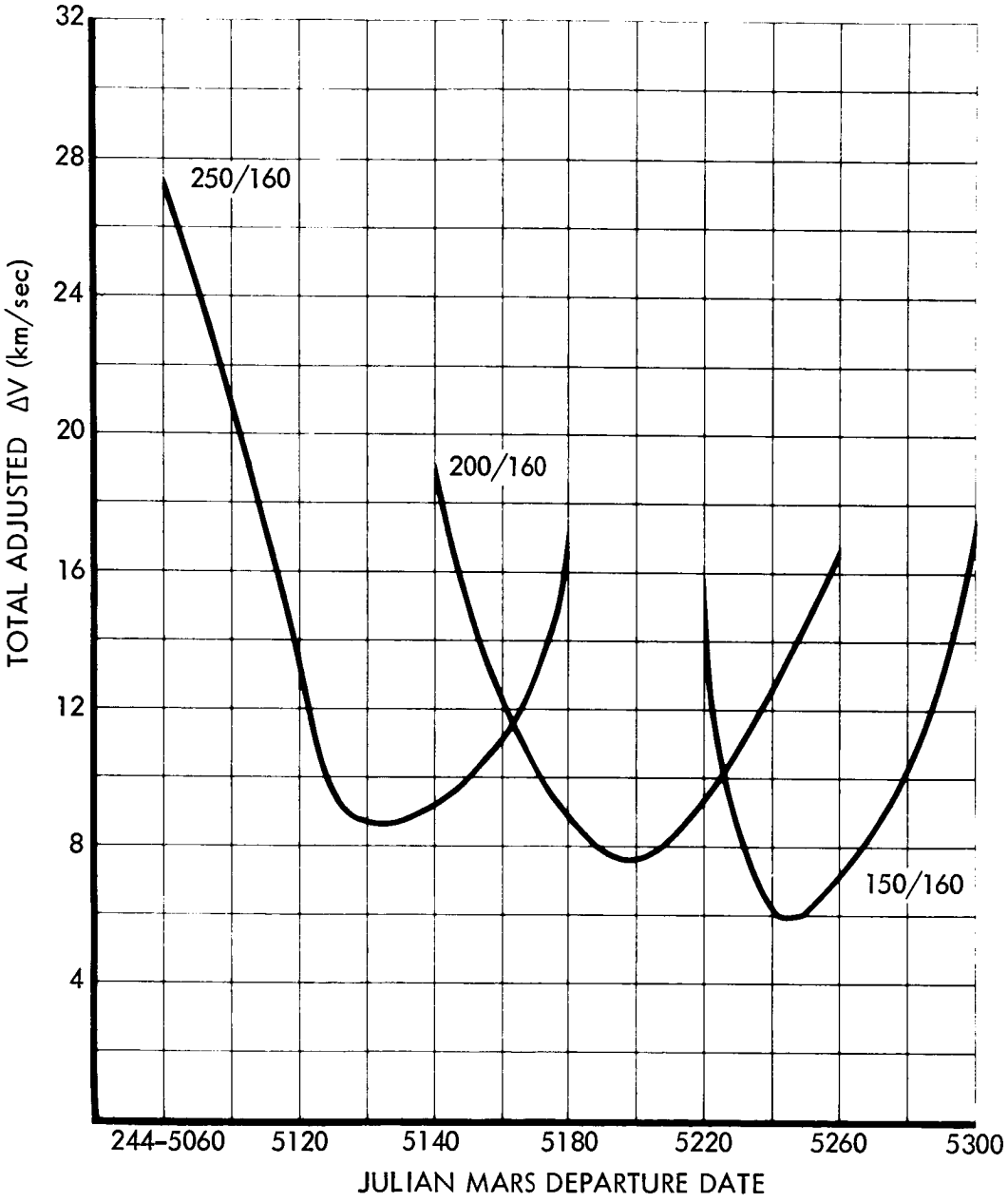


Figure 2.2-45: 1982 SWINGBY

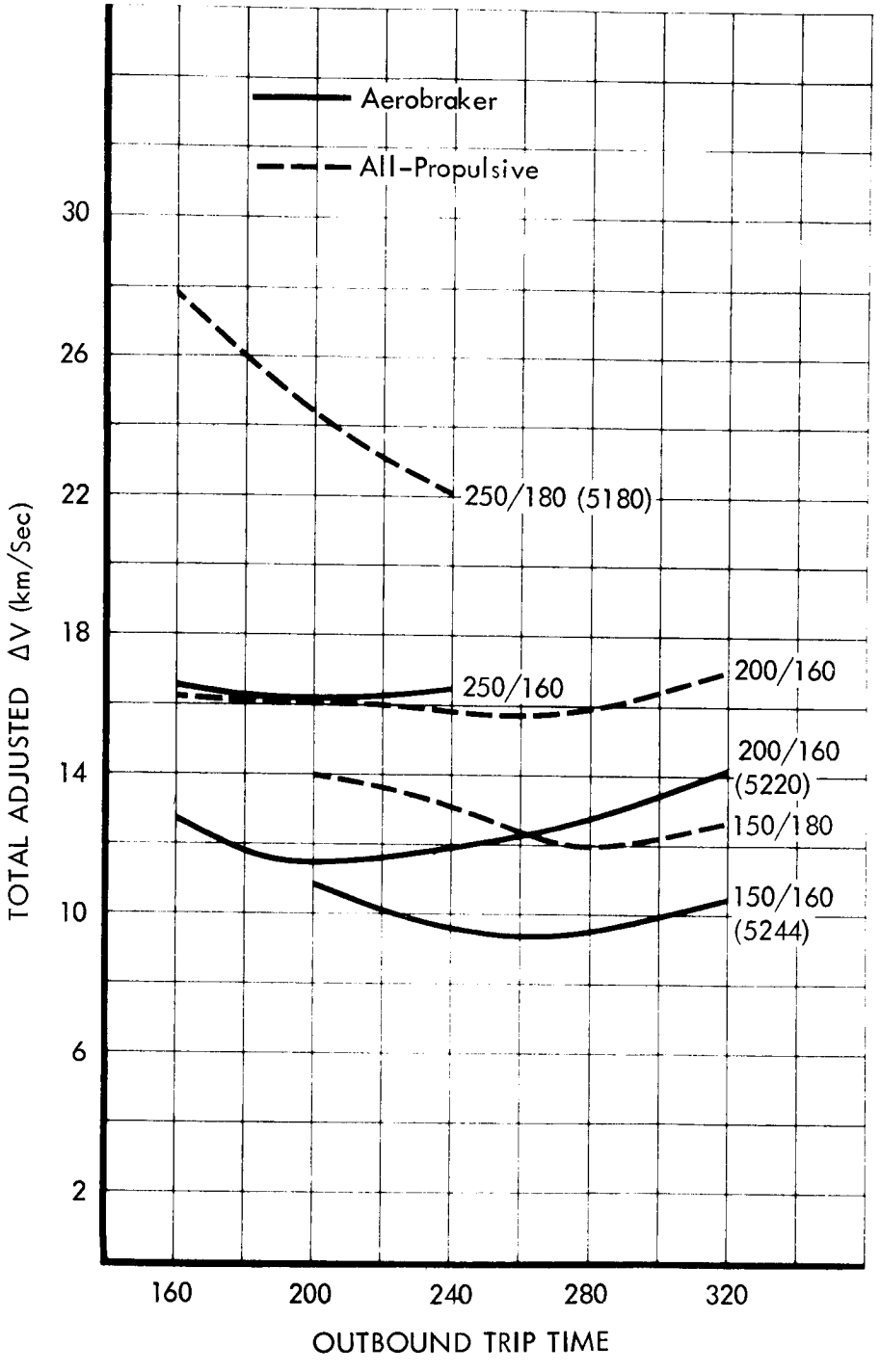


Figure 2.2-46: 1982 SWINGBY

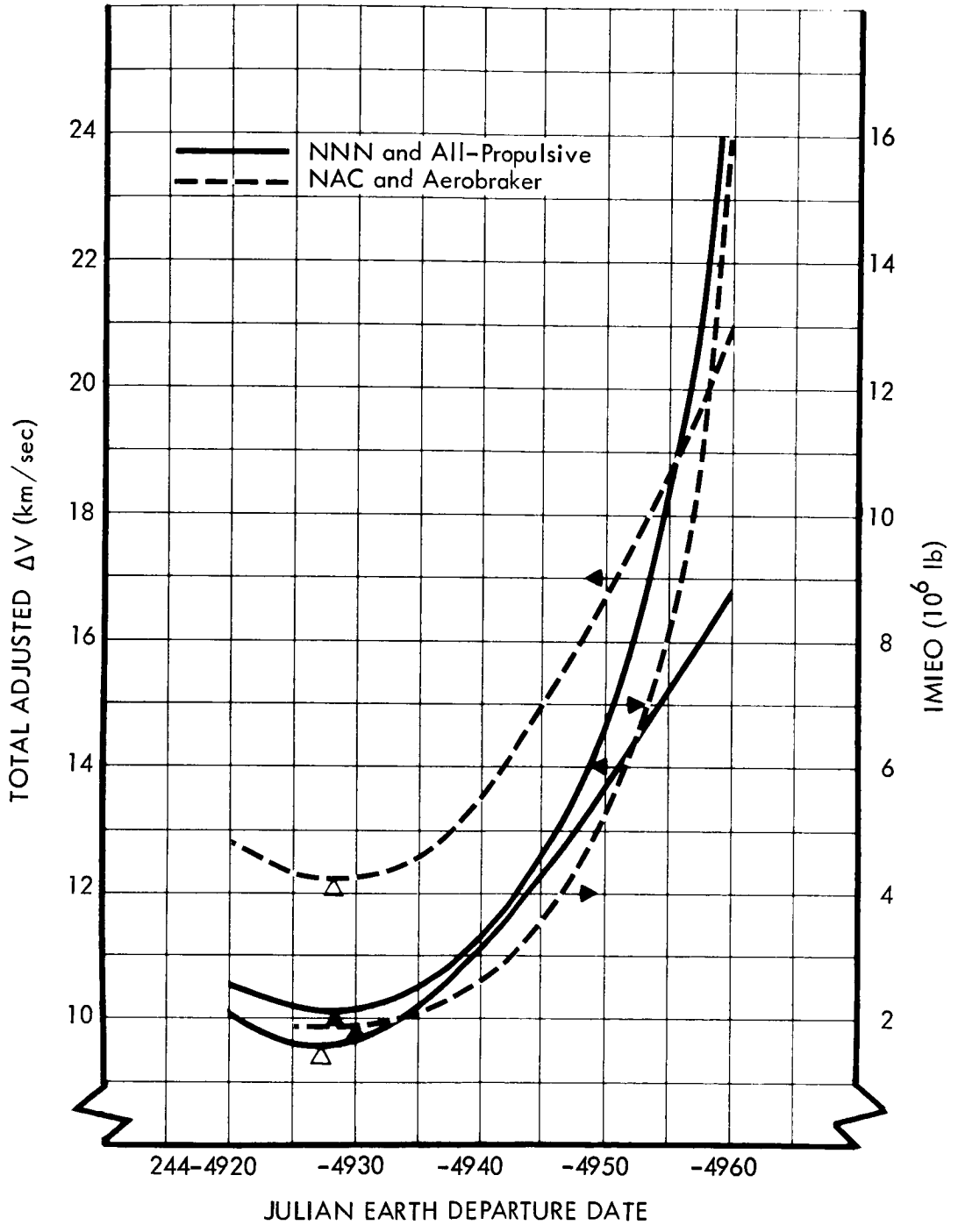


Figure 2.2-47: 1982 SWINGBY — 280/150/160 Mission

APPENDIX A-7

1980 VENUS WITH SHORT STAY MISSION CHARTS.
Figure 2.2-48 through 2.2-58. Explanation of these curves is given in Section 2.2.3.

Note:

The following notation is used to identify mission triptimes.

Outbound triptime (days) = XXX

$$\frac{\text{Total mission triptime (days)}}{\text{Outbound triptime (days)}} = \boxed{\frac{XXX}{XXX}}$$

Staytime (days) = 40

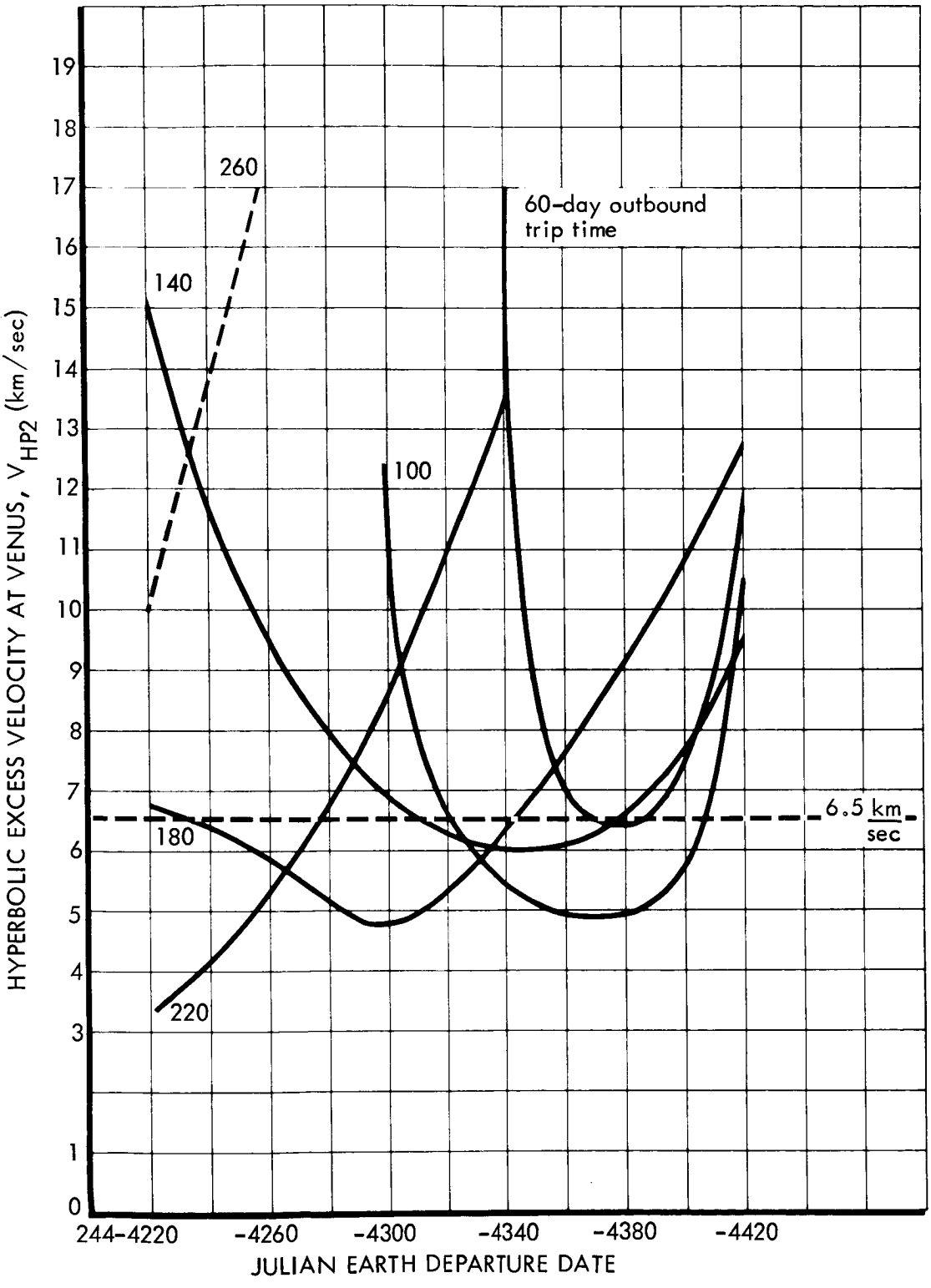


Figure 2.2-48: 1980 VENUS SHORT STAY — For Comparison Only

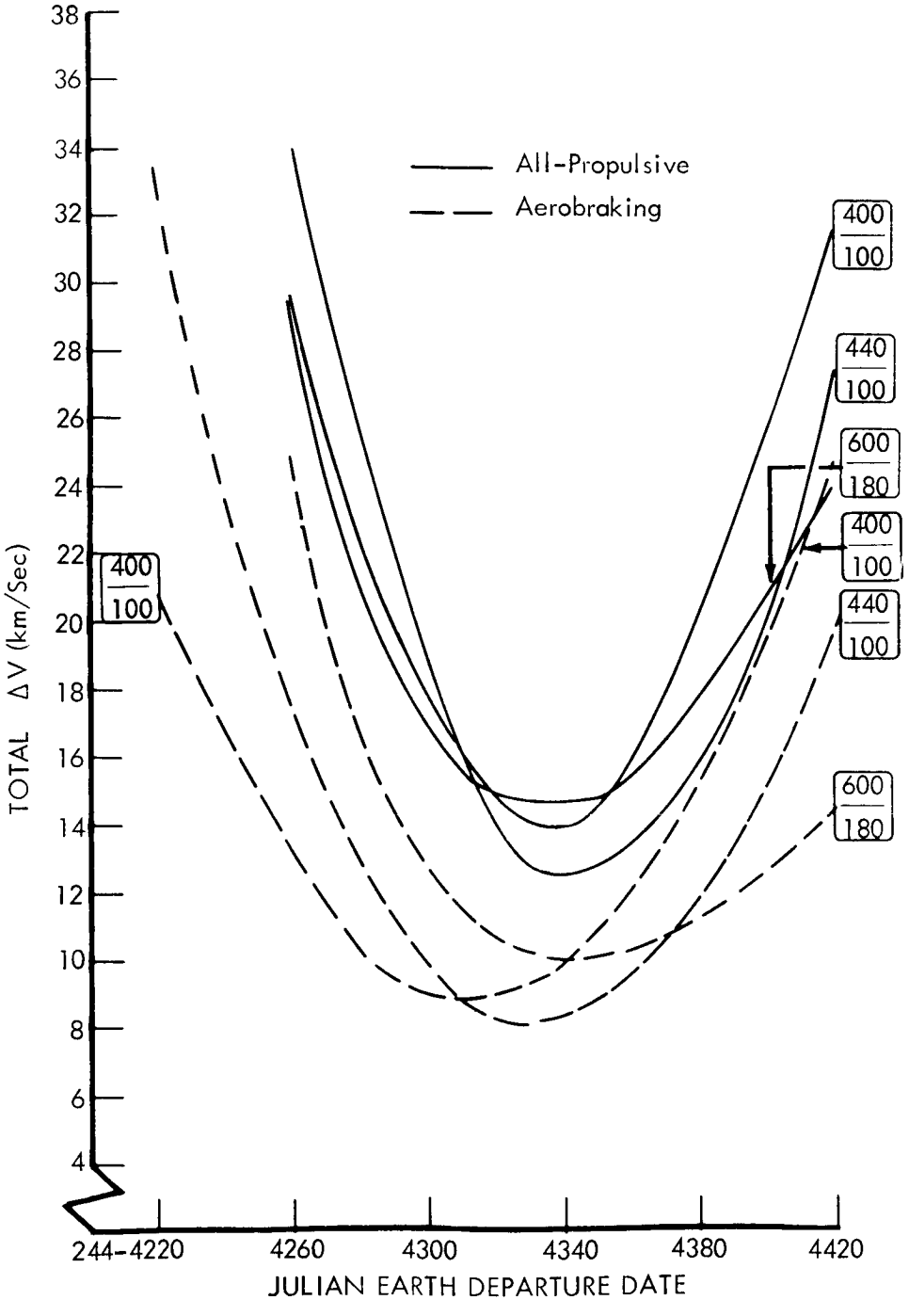


Figure 2.2-49: 1980 VENUS SHORT STAY

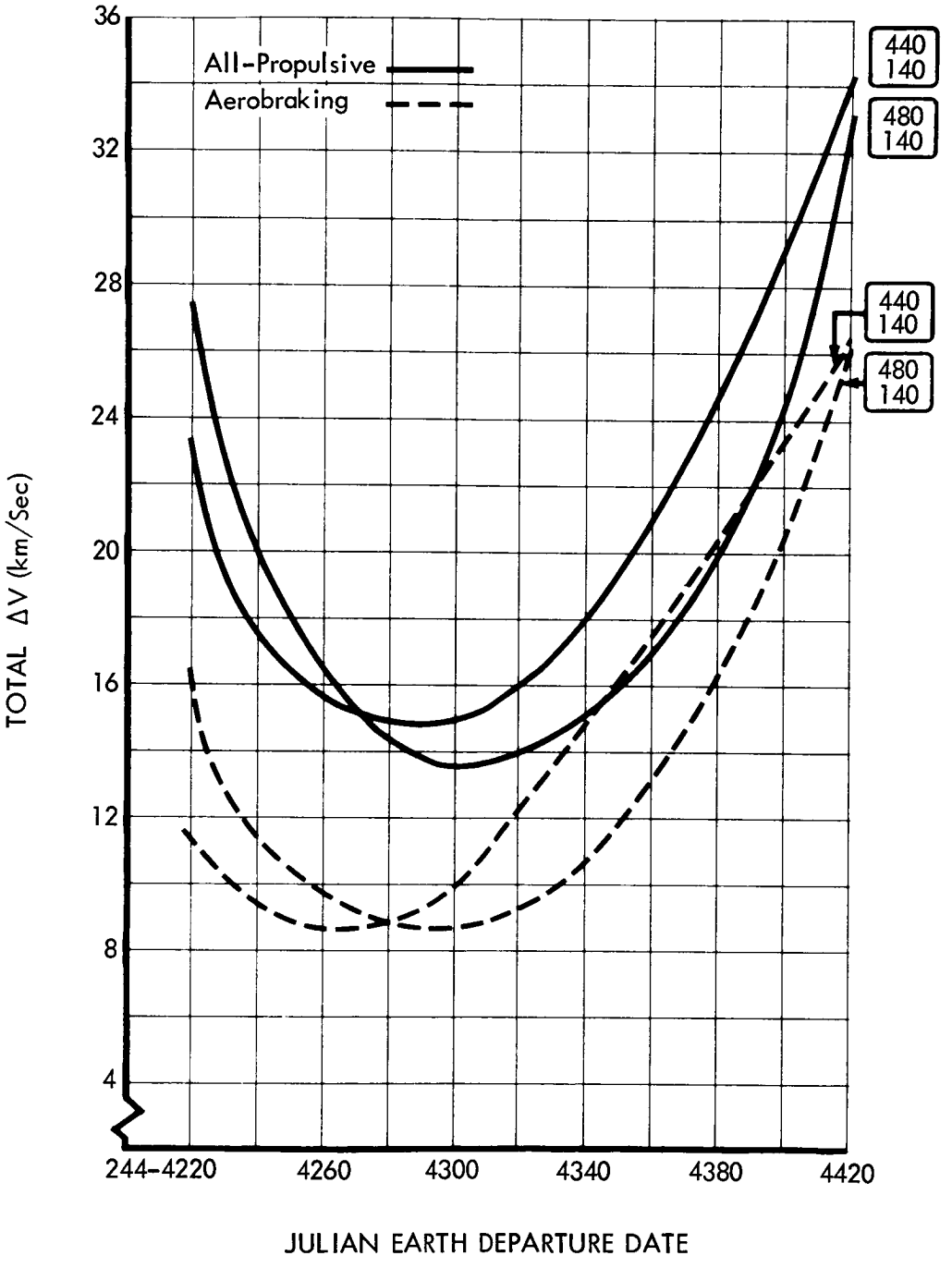


Figure 2.2-50: 1980 VENUS SHORT STAY

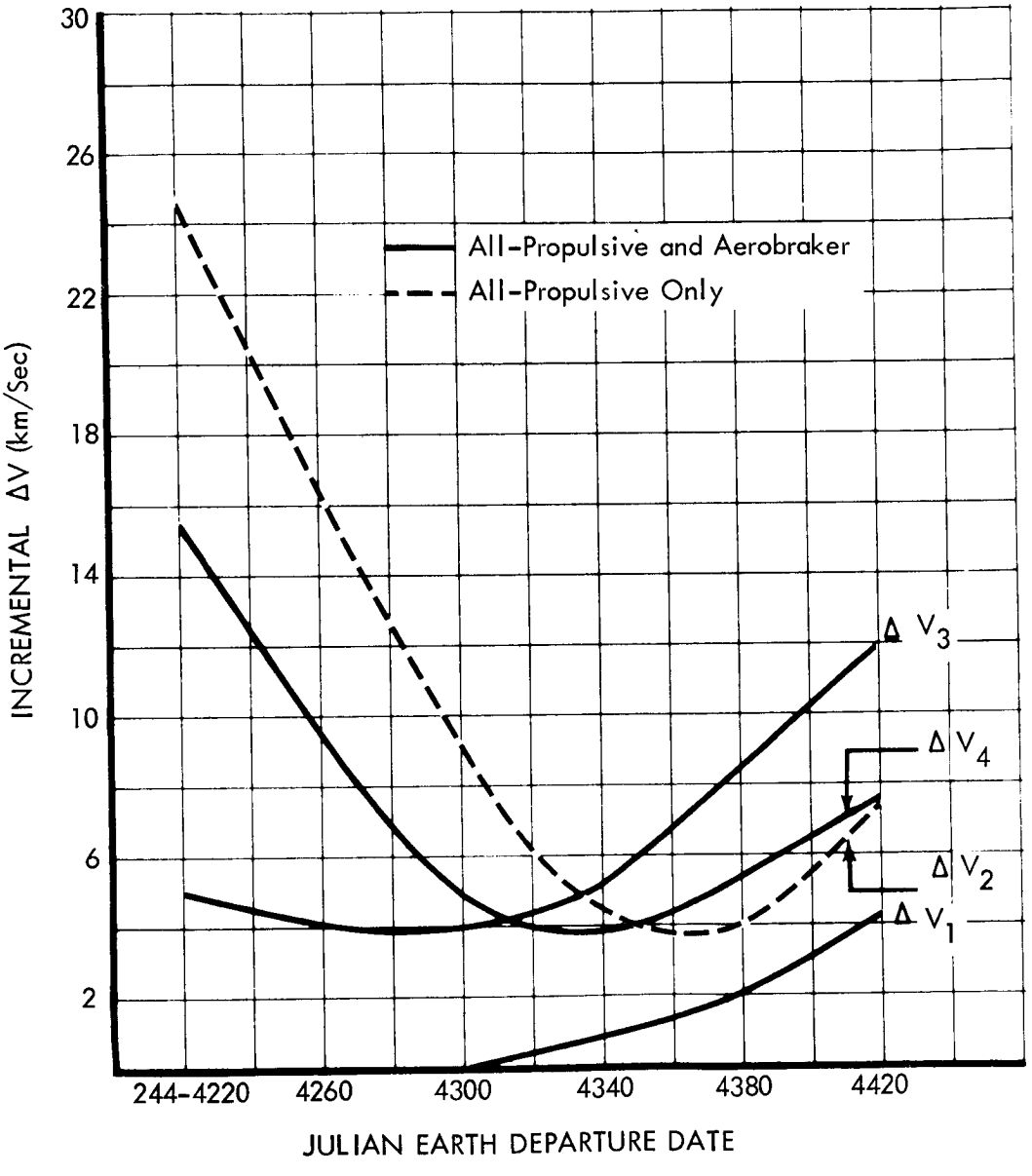


Figure 2.2-51: 1980 VENUS SHORT STAY — 400/100 Mission

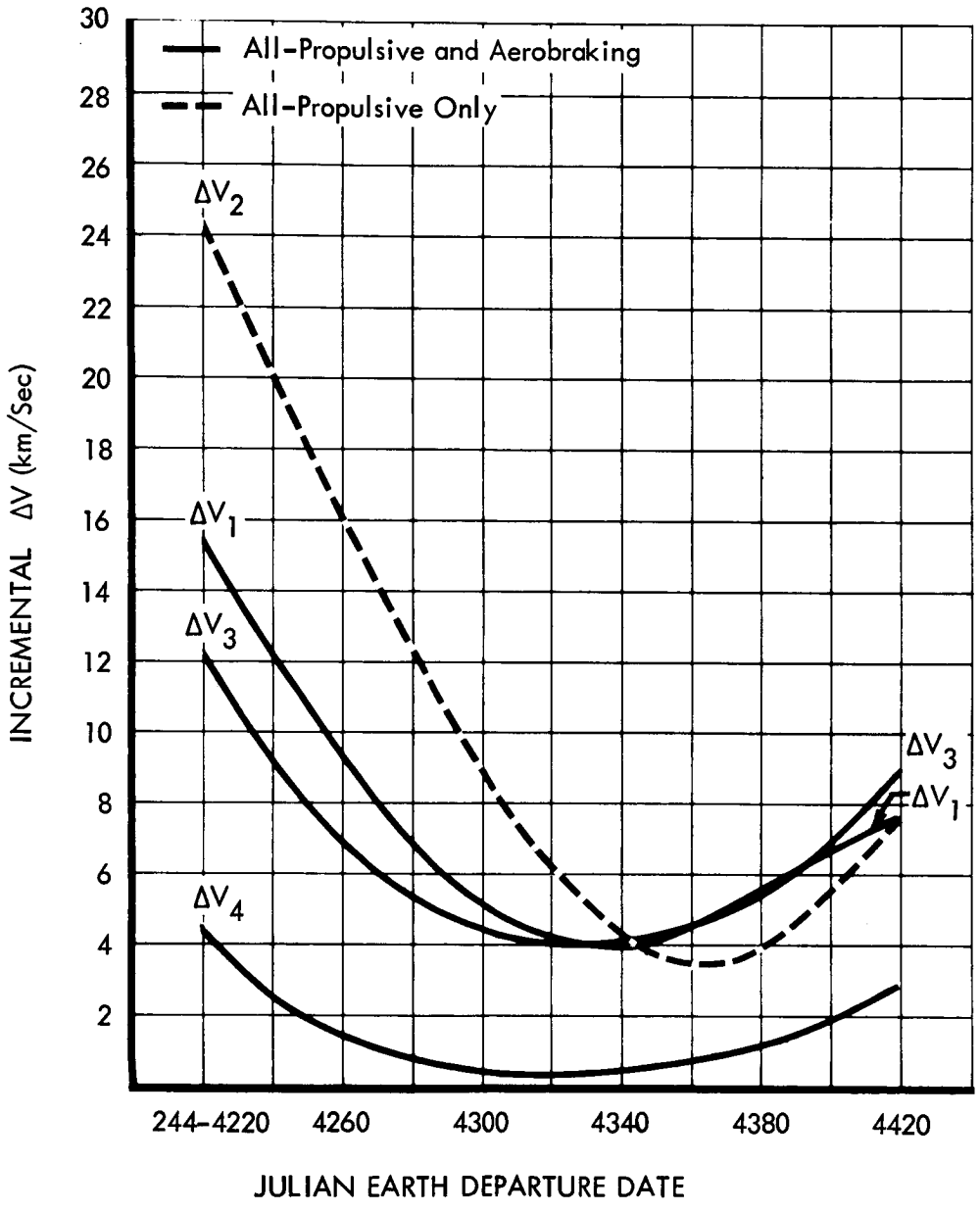


Figure 2.2-52: 1980 VENUS SHORT STAY — 440/100 Mission

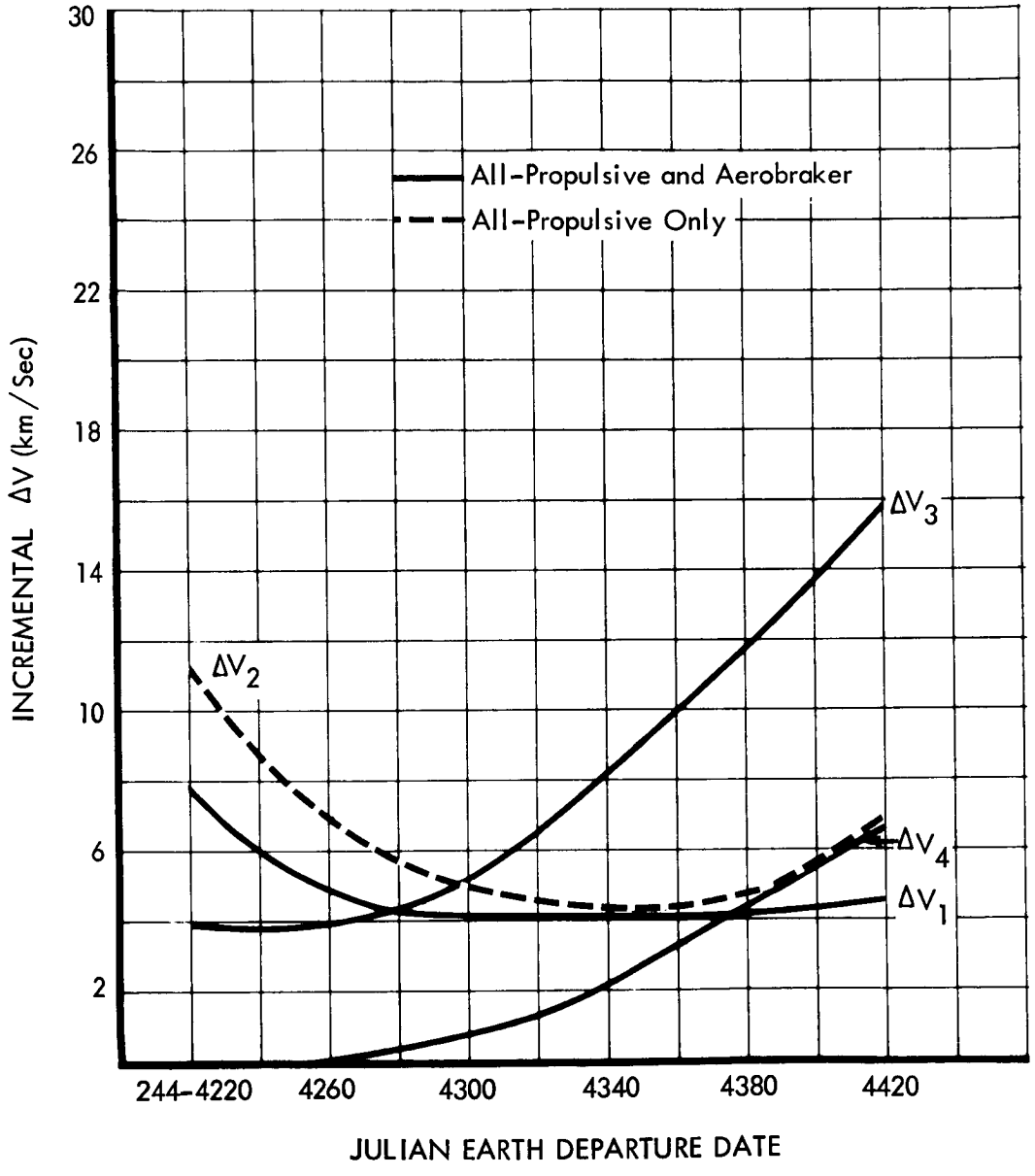


Figure 2.2-53: 1980 VENUS SHORT STAY — 440/140 Mission

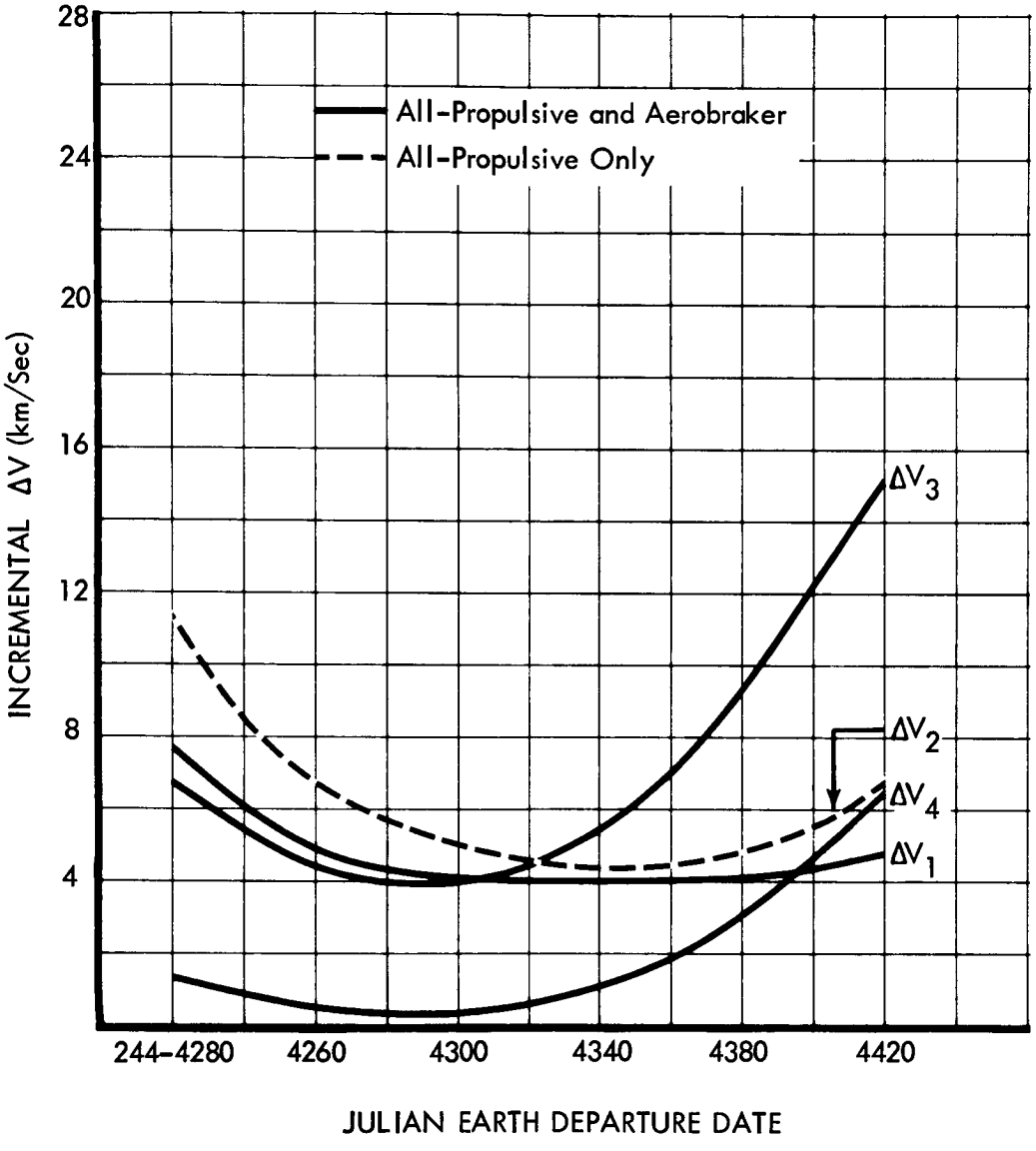


Figure 2.2-54: 1980 VENUS SHORT STAY — 480/140 Mission

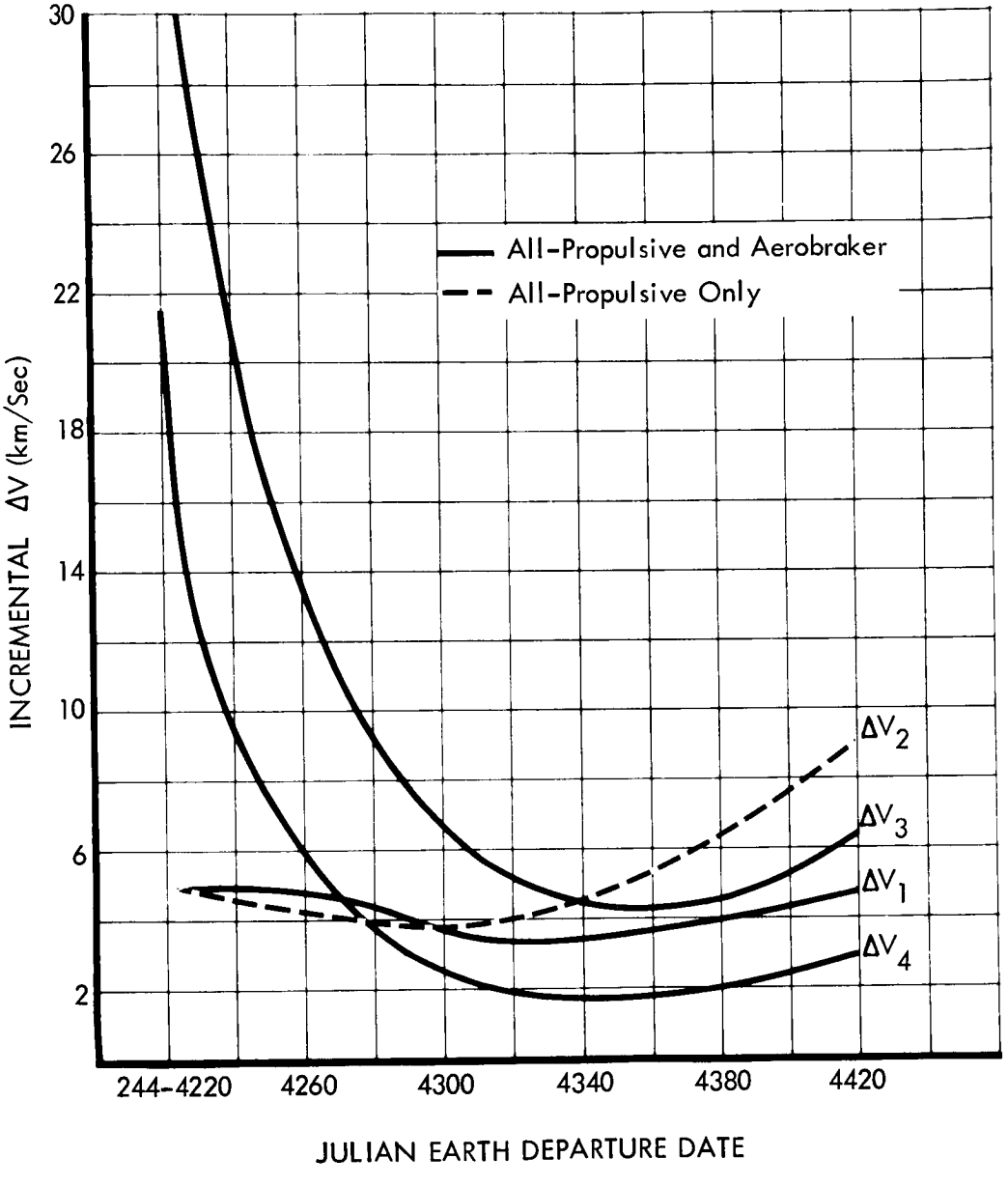


Figure 2.2-55: 1980 VENUS SHORT STAY — 600/180 Mission

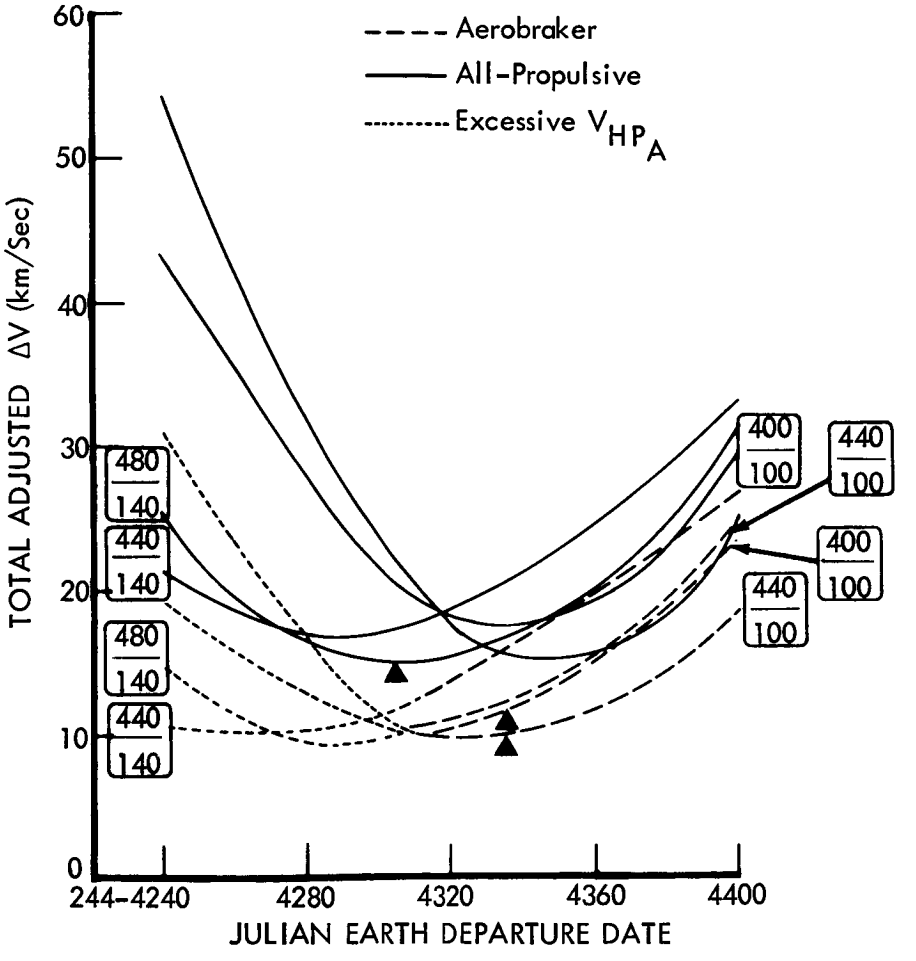


Figure 2.2-56: 1980 VENUS SHORT STAY

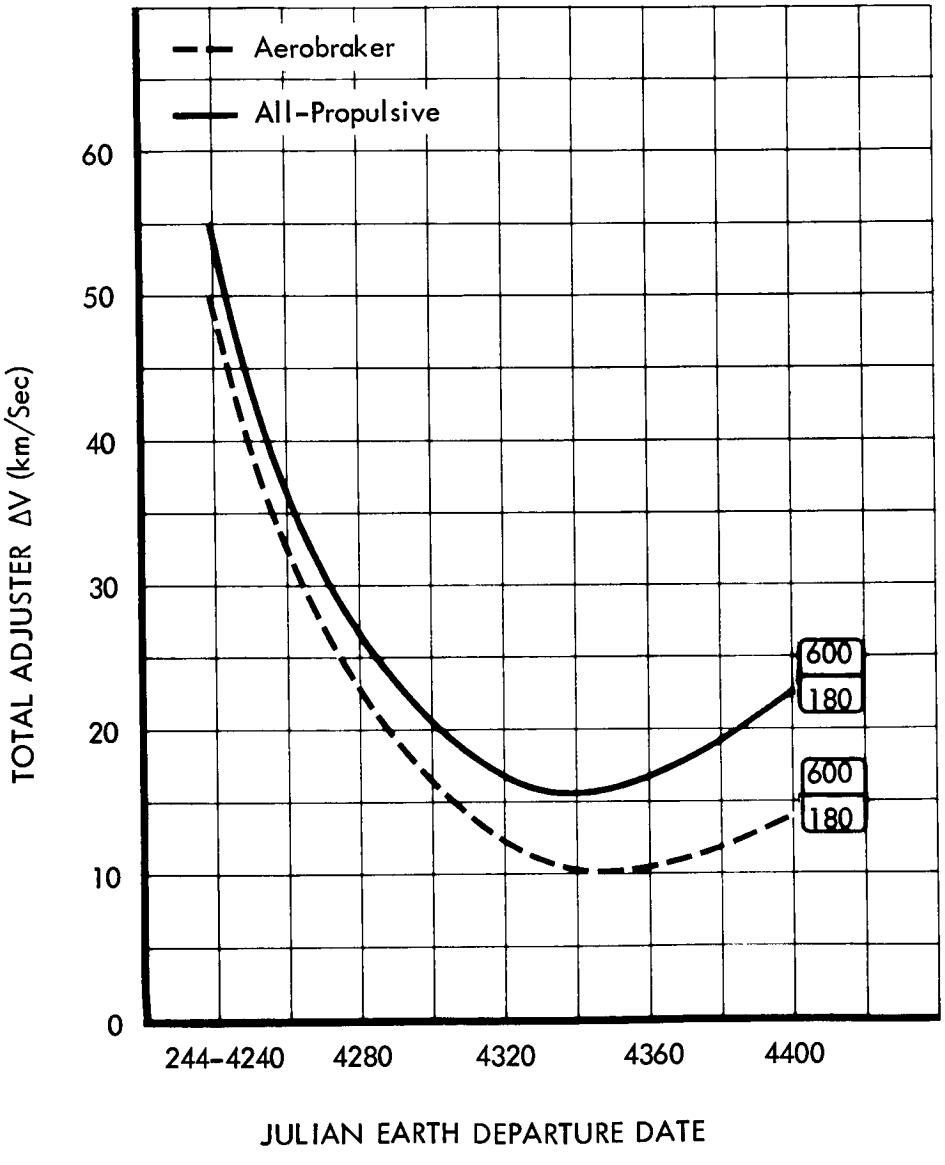


Figure 2.2-57: 1980 VENUS SHORT ATAY

APPENDIX A-8

1980 VENUS WITH LONG STAY MISSION CHARTS.
Figures 2.2-59 through 2.2-67. Explanation of these curves is given in Section 2.2.3.

Note:

The following notation is used to identify mission triptimes.

Outbound triptimes (days) = XXX

Outbound triptime/staytime/inbound triptime = XXX/XXX/XXX

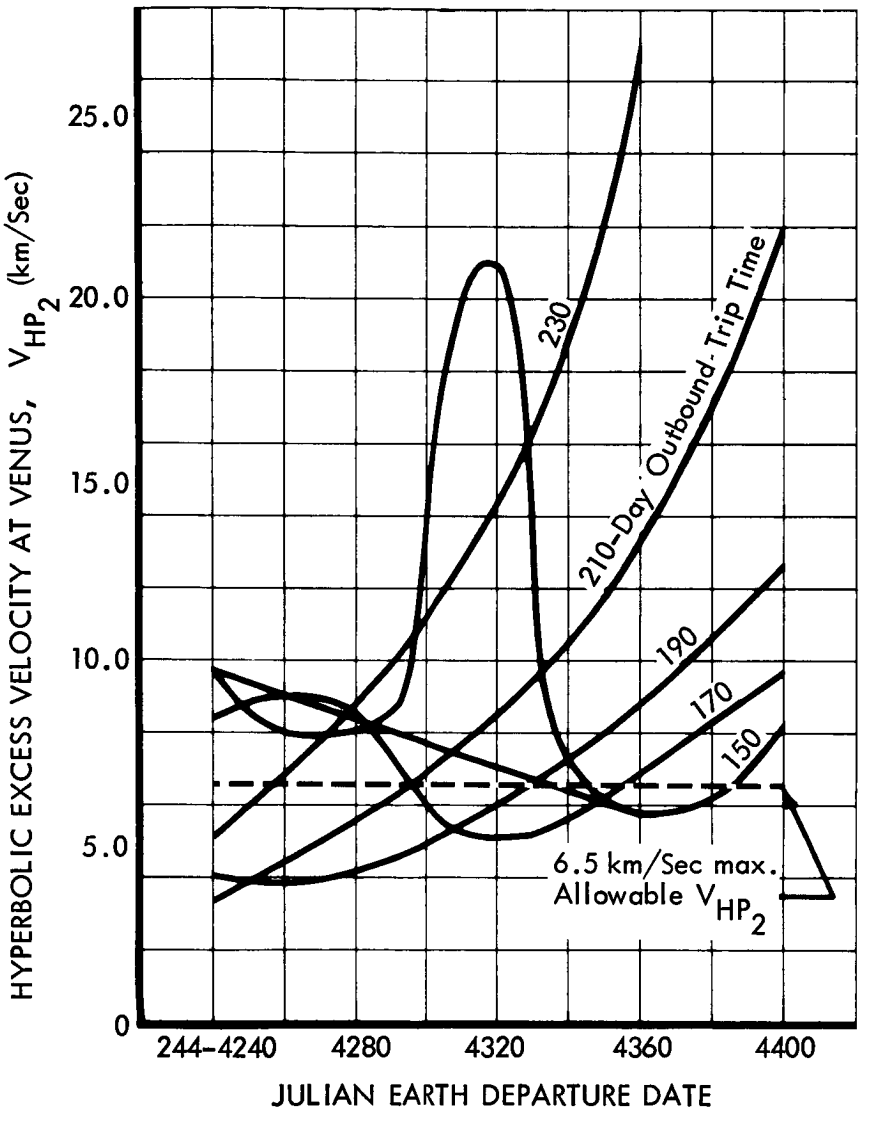


Figure 2.2-59: 1980 VENUS LONG STAY MISSION

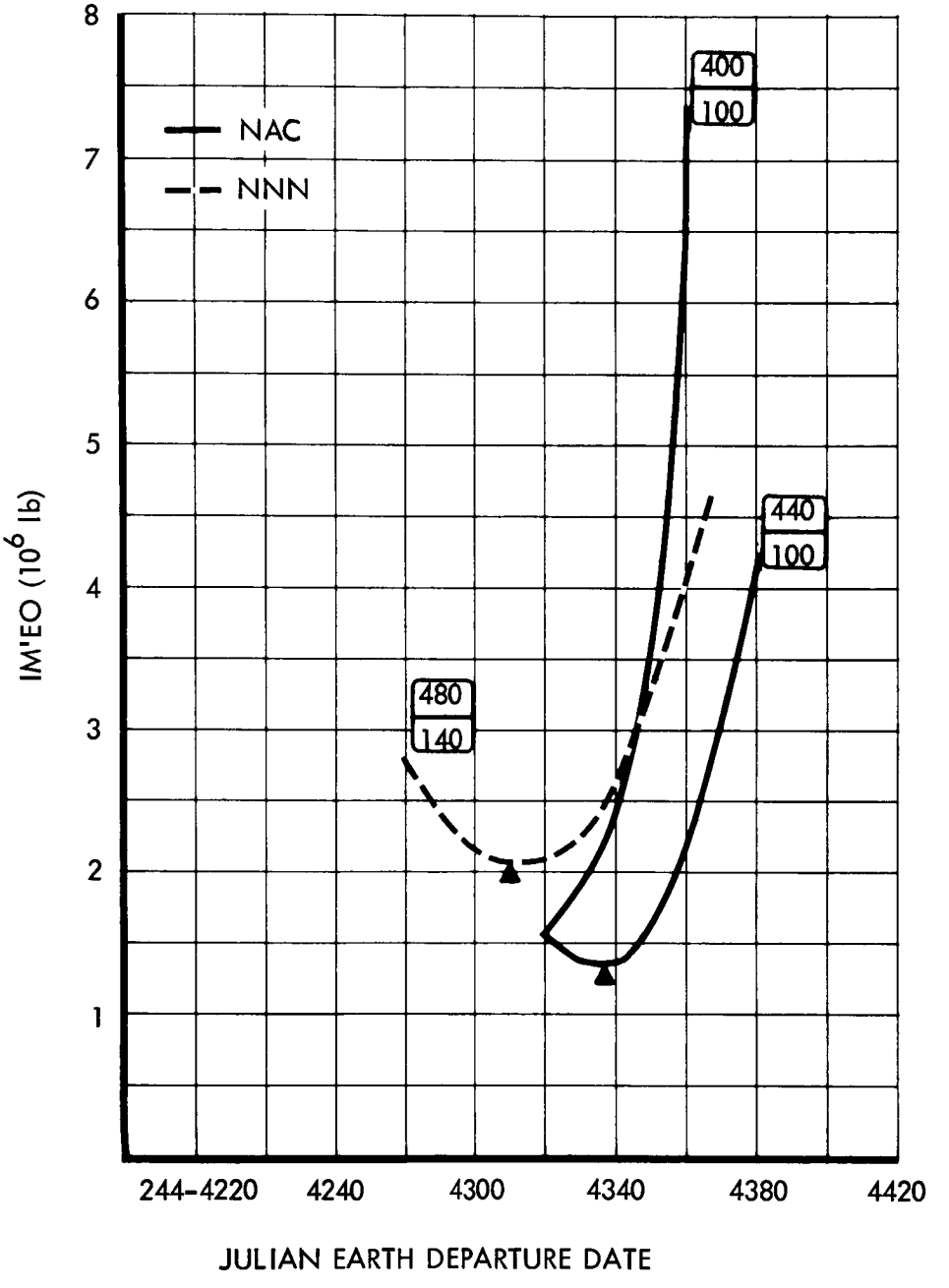


Figure 2.2-58: 1980 VENUS SHORT STAY

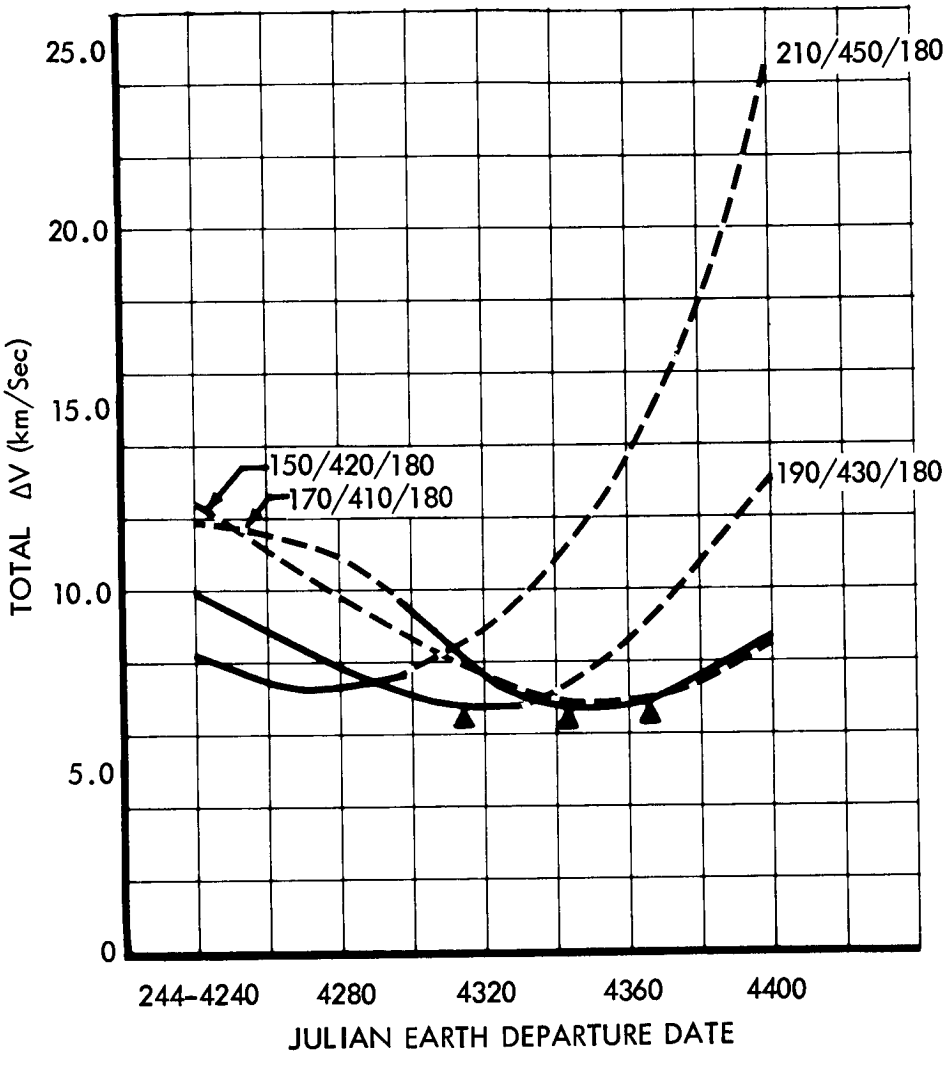


Figure 2.2-60: 1980 VENUS LONG STAY — Aerobraker

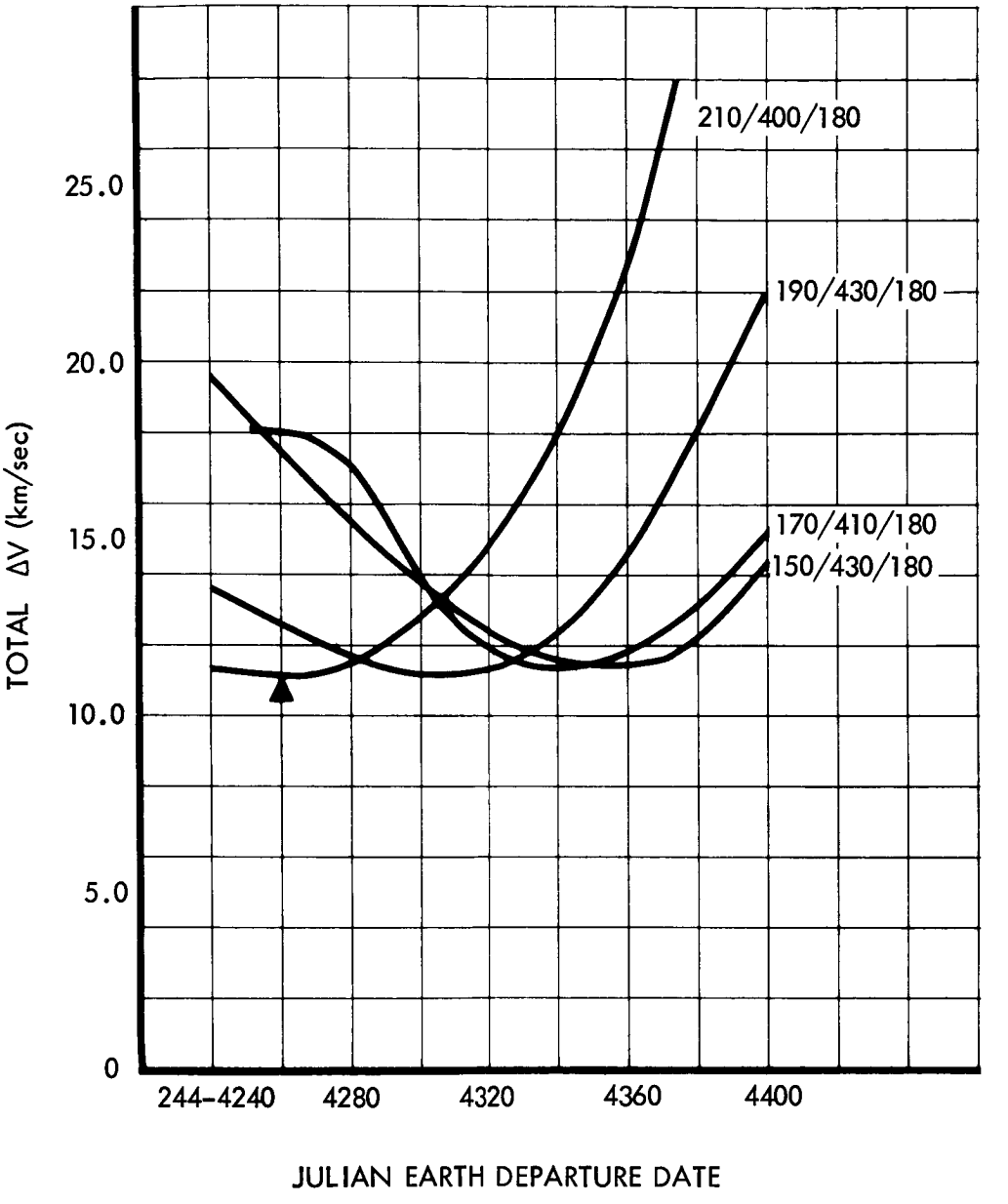


Figure 2.2-61: 1980 VENUS LONG STAY — All-Propulsive

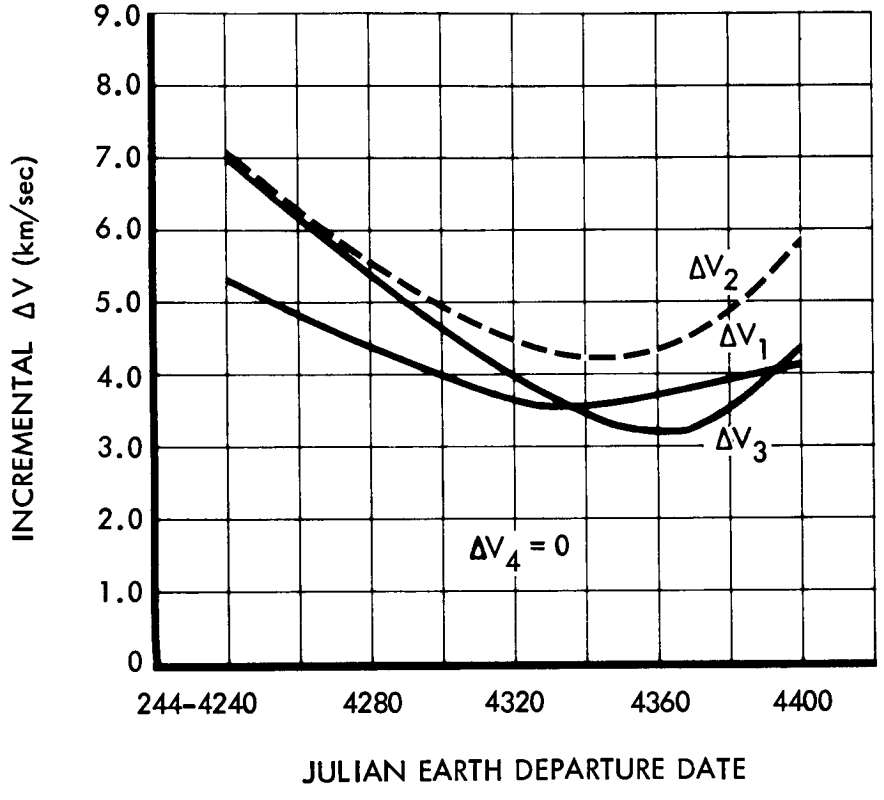


Figure 2.2-62: 1980 VENUS LONG STAY — 150/430/180 Mission

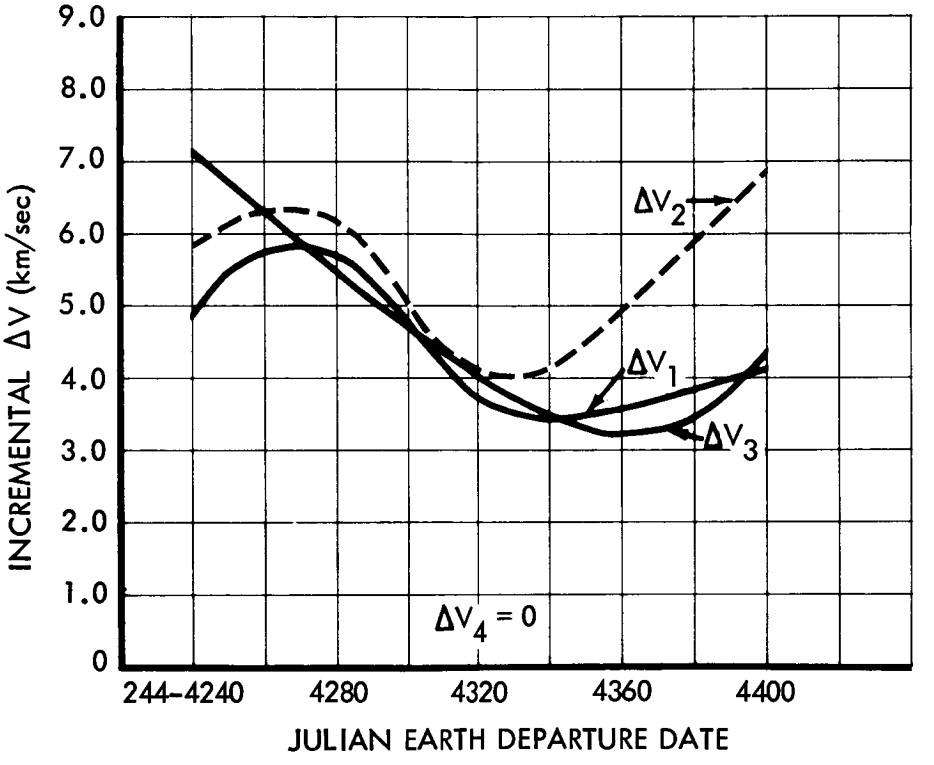


Figure 2.2-63: 1980 VENUS LONG STAY — 170/410/180 Mission

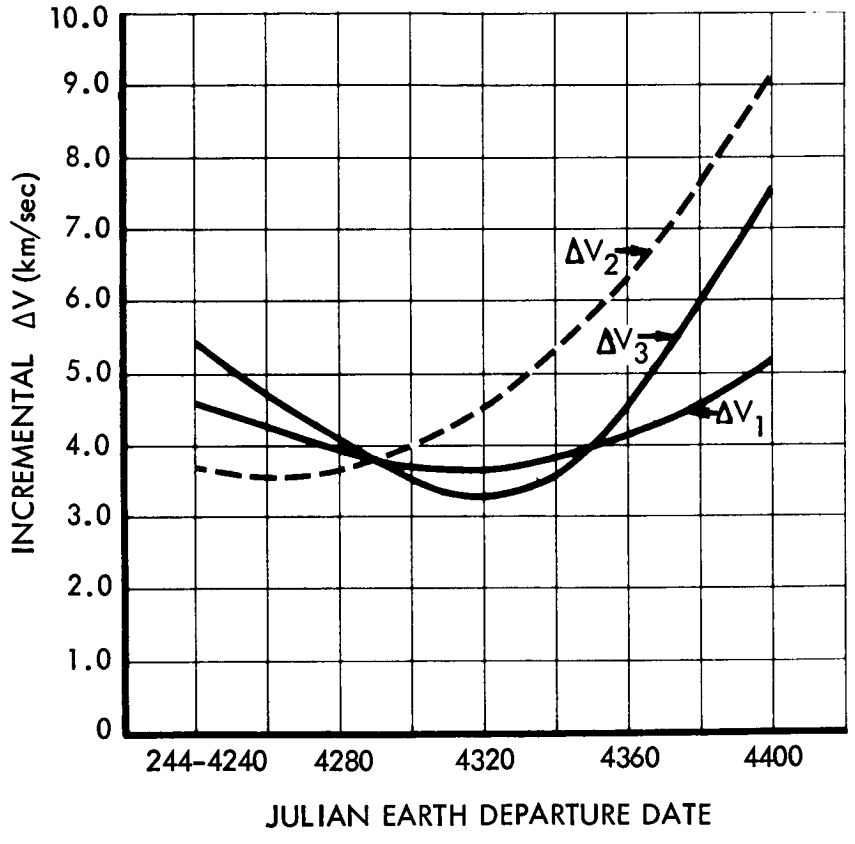


Figure 2.2-64: 1980 VENUS LONG STAY — 190/430/180 Mission

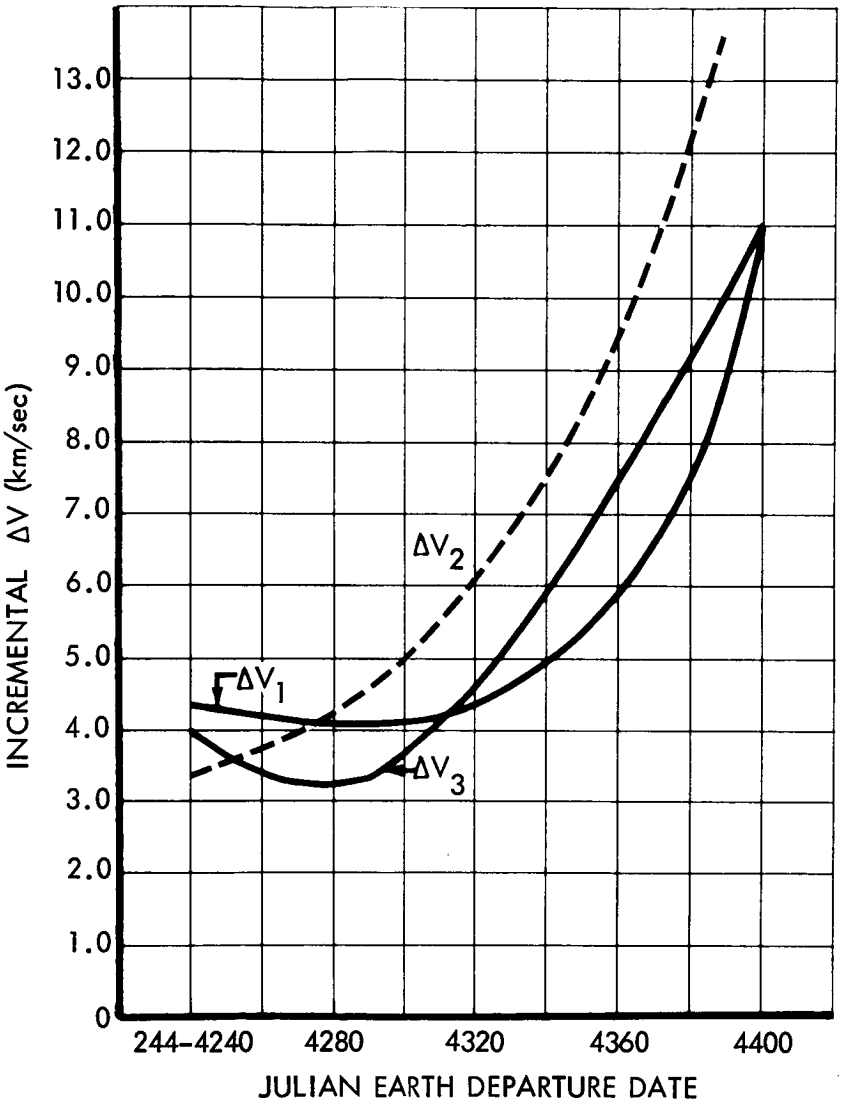


Figure 2.2-65: 1980 VENUS LONG STAY — 210/450/180 Mission

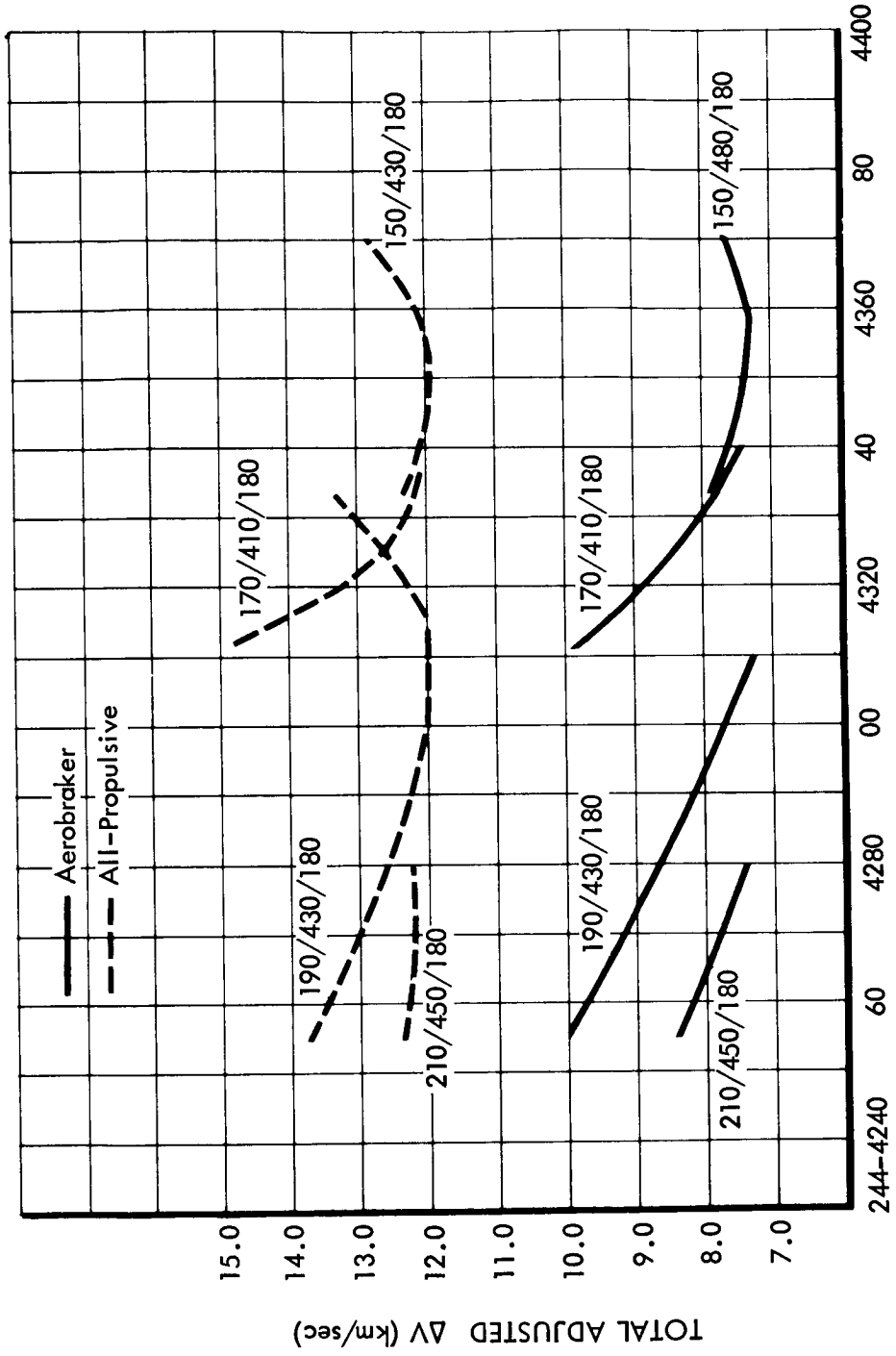


Figure 2.2-66: 1980 VENUS LONG-STAY

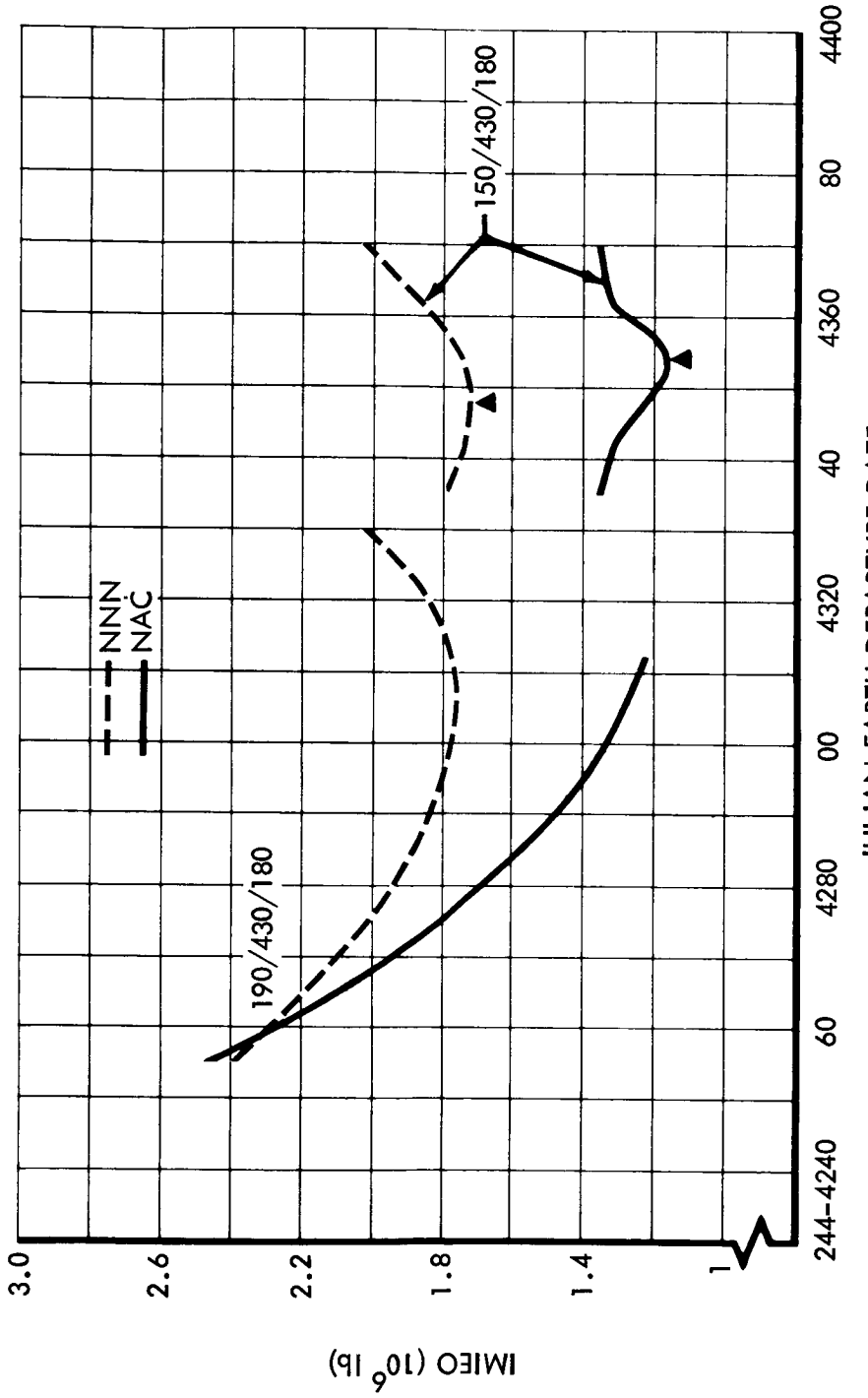


Figure 2.2-67: 1980 VENUS LONG STAY

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3.0 OPERATIONS

This section defines the basic operational plan for the selected missions and gives an assessment of the crew requirements. Operations are divided into mission and support operations. Mission operations cover all mission activities starting with Earth departure and ending with the touchdown of the Earth entry module (EEM) back at Earth. The support operations are divided into orbital support and Earth-based support. The orbital support operations cover all events starting with the Earth launch vehicle (ELV) launches from Earth and ending with the space vehicle ready for orbital launch and the assembly crews returned to Earth. The Earth-based support covers all operations from arrival of the space vehicle and ELV modules or components at the Cape through preparation of the ELV's for their Earth launching. Also included in the Earth-based support are the mission control and tracking operations throughout the entire mission and the final recovery support operations.

3.1 MISSION OPERATIONS

The approach used in this study to develop the operational plan for the mission operations was, first, to select an operational sequence and, second, to expand the plan, through a function and task analysis, to a level sufficient for definitions of crew requirements, program plans, and costs.

3.1.1 SELECTED OPERATIONAL SEQUENCE

The selected sequence of major operational events for the Mars opposition, conjunction, and swingby mission operations is shown in Figure 3.1-1. The sequence for the Venus short and long-stay missions is presented in Figure 3.1-2. As a basis for this sequence, the following decisions or assumptions were made:

1) The used nuclear stages will be separated and disposed of immediately following shutdown. Sufficient shutdown propellants will be provided, however, to allow for a short delay in the separation sequence.

2) Three midcourse corrections will be provided on each of the outbound and inbound legs of the trip. One restartable propulsion module will be used for outbound corrections and one for inbound corrections.

3) All excess mass will be discarded prior to each propulsive maneuver. No significant mass diaposal will occur between midcourse corrections.

Representative time milestones are shown in Figure 3.1-1 for four Mars missions in terms of days from the Earth orbital launch. The same is shown in Figure 3.1-2 for two Venus missions. Figure 3.1-3 provides better visualization of the major mission operations events. The mission is initiated by the disposal (in Earth orbit) of the meteroid shield of PM-1. The space vehicle is then launched into the interplanetary transfer trajectory with PM-1. Three midcourse corrections, occurring

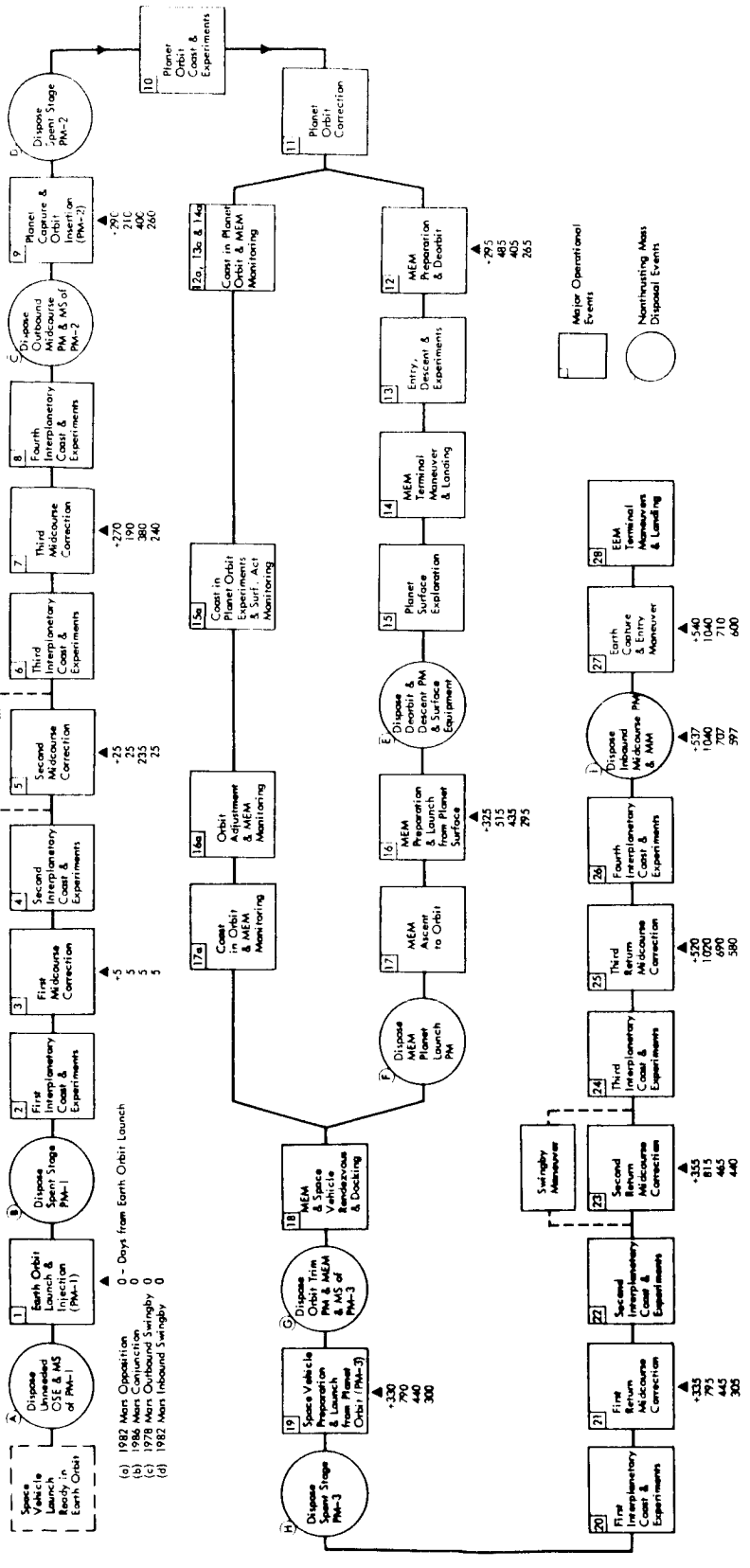


Figure 3.1-1: MISSIONS OPERATIONS SEQUENTIAL FLOW MARS OPPOSITION, CONJUNCTION, AND SWINGBY MISSIONS

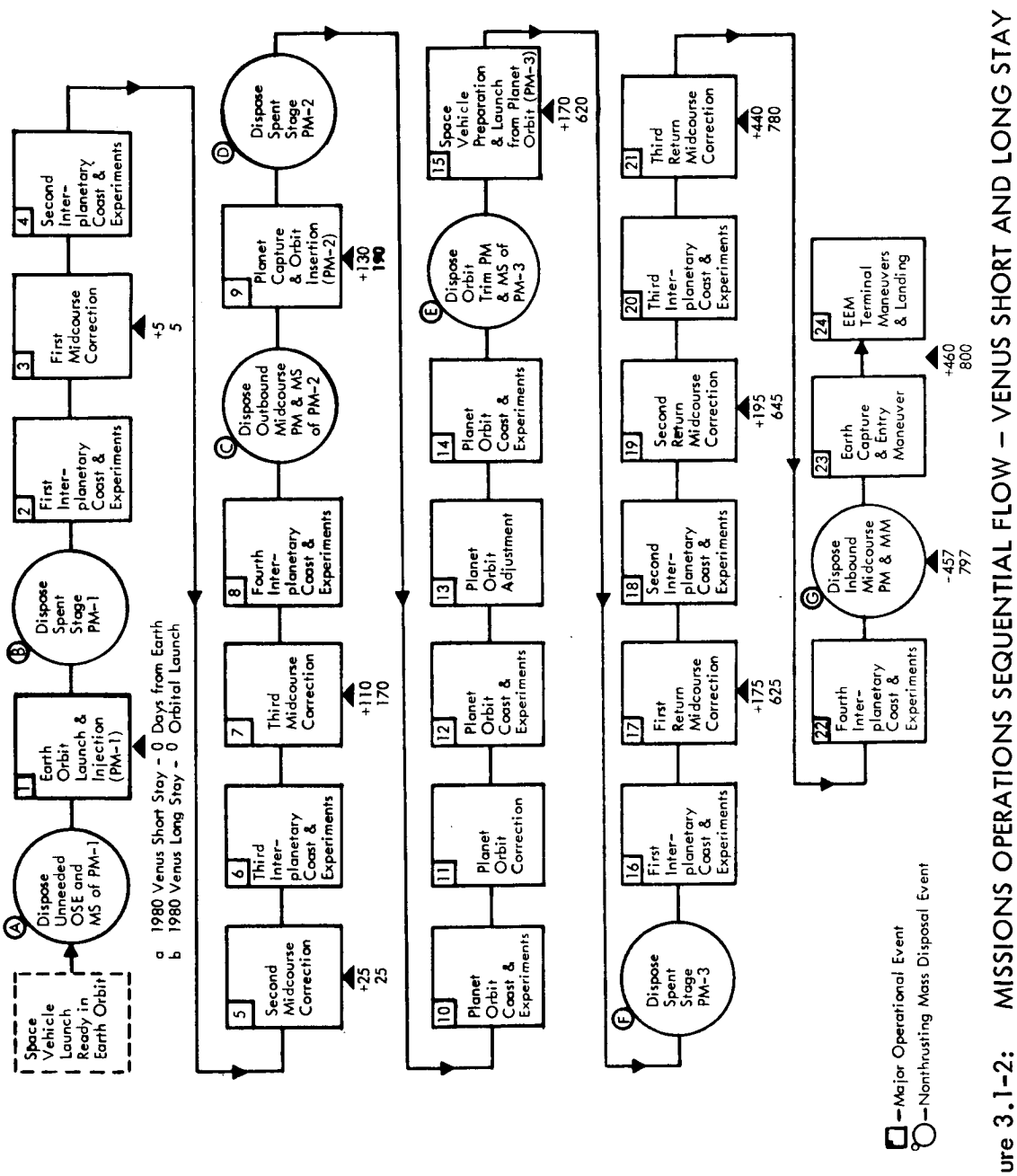


Figure 3.1-2: MISSIONS OPERATIONS SEQUENTIAL FLOW – VENUS SHORT AND LONG STAY MISSIONS

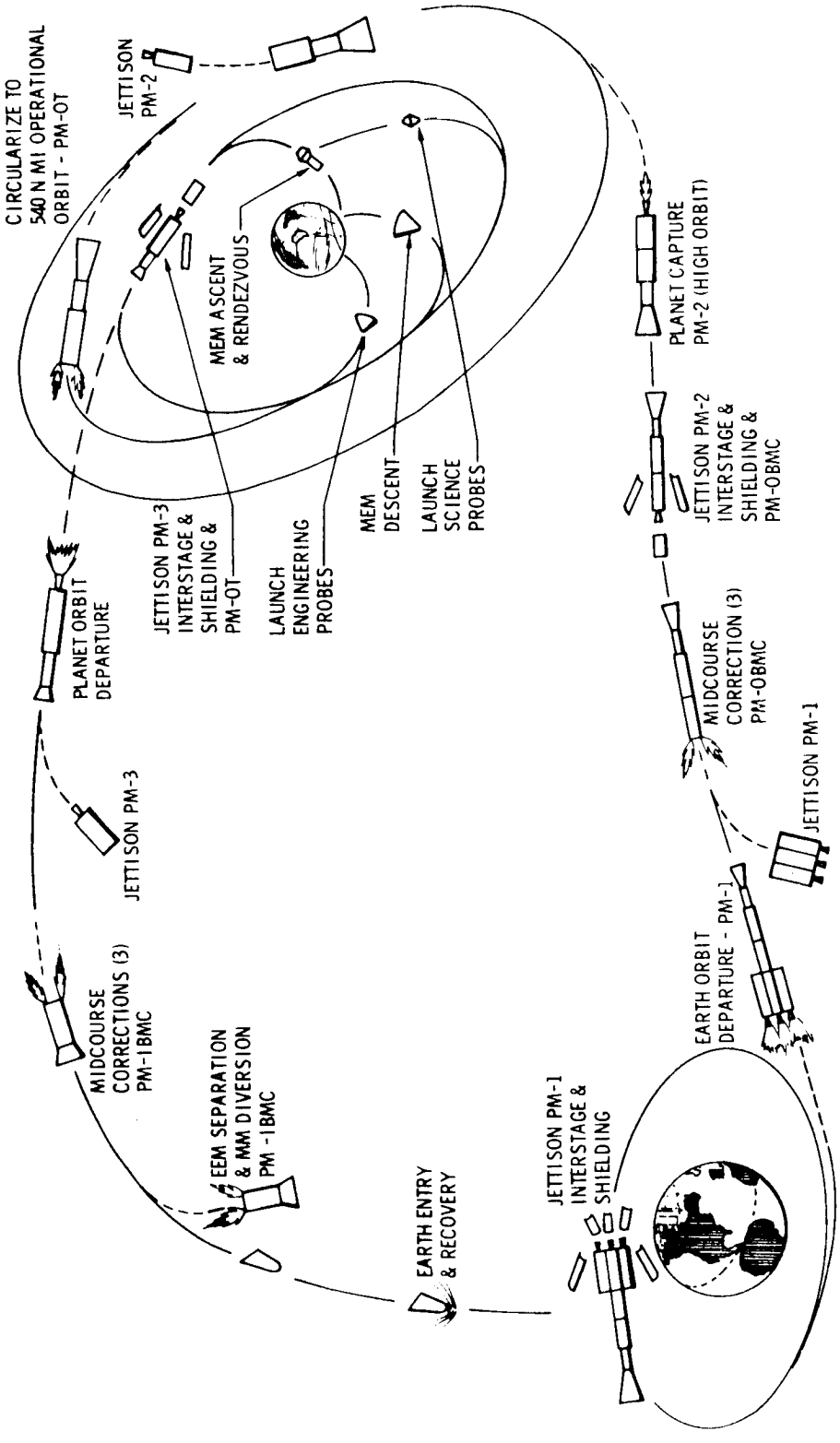


Figure 3.1-3: MISSION EVENTS SEQUENCE

at 5 and 25 days after orbital launch and 20 days prior to arrival, are assumed for each interplanetary leg. The planet capture and insertion into a high altitude orbit (1200 km) is made using PM-2, and the space vehicle descends to its lower operational orbit (1000 km circular) using orbit trim propulsion. The spent PM-2 stage is separated and left in the higher orbit. A 40-day stay at the planet is assumed except for the Mars conjunction class and Venus long-stay missions. Approximately 30 days of manned surface exploration are assumed along with 40 days of orbital experiments, including probes. Upon completion of the orbit and surface experimentation and disposal of used modules and hardware, the orbital departure is accomplished by a PM-3 stage. Within the final 3 days prior to Earth entry, the crew is transferred to the EEM, along with equipment, data, and samples to be returned, the mission module (MM) and other used modules are discarded, and a direct capture and entry is accomplished by the EEM. The operational sequences of the Venus missions do not include manned descents, but do involve probe exploration of the atmosphere and surface as a part of the experiment program. In the Venus swingby missions, the swingby maneuver is shown in the figure as occurring at the time of the second midcourse correction, but this does not eliminate the requirement for that correction.

3.1.2 FUNCTION AND TASK ANALYSIS

The function and task analysis was performed for a 1982 Mars opposition mission. The analysis expanded the major operational events into descriptions of the tasks required to perform each event and a listing of general procedures used to accomplish the tasks. From these, personnel, time, and equipment requirements were derived. For each task, the number of persons required to perform the task was estimated and the necessary skills were defined. An elapsed time was estimated for each task and the required man-minutes were calculated. The primary pieces of equipment involved in the task's performance were identified and the task location was stipulated. Function and task analysis summary sheets are included as sheets 1 through 16 of Figure 3.1-4. The results are integrated with the results of experiment and maintenance analyses in Section 3.3.

3.2 SUPPORT OPERATIONS

Support operations are divided into orbital and Earth-based support and include all operations required to accomplish the missions except the mission operations themselves. This section contains a representative support operations plan for program planning and costing.

3.2.1 ORBITAL SUPPORT

The orbital support operations cover all of the operational events from Earth launch of the space vehicle modules up to the point at which the vehicle is assembled and checked out and is ready for orbital launch. The selected orbital support sequence is described below.

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FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
Space Vehicle Launch Ready in Earth Orbit										This is not an event; it describes the state of the vehicle at that time.
Ⓐ Dispose Unnecessary Orbital Support Equipment (OSE) and Meteoroid Shield (MS) of PM-1 (1 hour)	1. Separate Meteoroid Shield from PM-1.	1. Activate release mechanisms for PM-1 meteoroid shield. 2. Initiate MS and INS separation from space vehicle.	1	FO	30 min	30	90	Release mechanisms and separation systems for meteoroid shield.	Around PM-1 and in general area around space vehicle. Primary control exercised from MM control console.	This event may be considered one of the final orbital support operations.
	2. Separate Unnecessary OSE from Space Vehicle.	1. Assure proper release of all OSE connections with the space vehicle. 2. Separate OSE from space vehicle and assemble with MS of PM-1. 3. Transport OSE and MS to safe distance. Retain for reuse prior to injection if required.	2	FO and Gen	30 min	60		Adequate numbers of conveniently located outlets or connect points on space vehicles for OSE utilization. Keeping station mod for OSE and MS and INS could be manned or unmanned.		
1 Earth Orbit Launch and Injection (PM-1) (3.5 hours)	1. Final Countdown	1. Verify space vehicle's spatial position and synchronize appropriate parameters with DSIF for future tracking and navigation support. 2. Coordinate orbital launch requirements with ground control and update on-board computers. 3. Final on-board subsystems checkout and abort systems readiness tests. 4. Preparation for engine ignition.	3	FO, N&G and Gen	1 hr	180	575	On-board navigation systems for near planet and heliocentric position determination. On-board computers for navigational support. Communications system. Readily available abort systems.	Mission module (MM) and abort stations.	
	2. Engine Ignition	1. Initiate manual engine startup sequence. 2. Monitor instrumentation and verify satisfactory reactor and propellant systems control.	1	Prop	5 min	5		Flight control console in the MM with abort warning and control provisions in both MM and EEM.		During all propulsion maneuvers except attitude change or other comparable maneuvers, all crew members should be in their pressure suits and all members not involved in the maneuver control should be located at their abort stations.
	3. Perform Propulsive Maneuver	1. Initiate programmed power buildup sequence. 2. Assure satisfactory reactor and propellant systems operation. 3. Assure prescribed vehicle attitude during thrusting. 4. Assure achievement of required incremental velocity change and prepare for engine shutdown.	2	FO and Prop	30 min	60		Manual and automatic (computer programmed) propulsion system and guidance control with built-in trouble shooting, feeback circuitry, and information display for PM-1.		
	4. Engine Shutdown	1. Initiate programmed engine shutdown sequence. 2. Assure satisfactory power delay and after cooling. 3. Adjust engine and propellant control operations to a prescribed stage-separation mode.	2	FO and Prop	2 hr	240		Adequate propellant quantities and control systems to accomudate nominal and emergency after cooling prior to stage separation		Engine shutdown for the nuclear stages will require aftercooling which provides a decaying thrust that should be included in the thrust program. Cooling down until T/W is approximately 0.001 should facilitate an adequate state separation. If stage is to be retained added cooling is needed.

Typical mission assumes nuclear PM-1, PM-2 and PM-3 and chemical (LO_2/H_2) PM-4.

Figure 3.1-4: B TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS

Fig 3.1-4:A 145

FOLDOUT FRAME

FOLDOUT FRAME

~~145~~ & 146

FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
(1) Dispose Spent Stage PM-1. (0.5 hour)	1. Release Stage Attachment.	1. Activate release mechanisms. 2. Assure complete release.	3	FO, Prop and Gen	30 min	90	90	Positive stage release and separation systems.	Mission module (MM) and abort stations	
	2. Separate PM-1.	1. Assure PM-1 is in separation mode of operation. 2. Activate separation systems. 3. Assure adequate separation trajectory and relative orientation between PM-1 and the space vehicle.						Attitude control and maneuvering capability must be provided on either PM-1 or the space vehicle to ensure proper separation.		An error in the relative orientation or the separation trajectories of the PM-1 stage and space vehicle could result in a severe, if not fatal, radiation hazard.
[2] First Interplanetary Coast and Experiments (5 days)	1. Space Vehicle Operations a. Systems Monitoring and Control	1. Continuous monitoring of command console. 2. Communicate with Earth-based control and exchange pertinent mission information. 3. Maintain mission log. 4. Perform periodic subsystems readiness and performance tests. 5. Initiate periodic malfunction alert and mission abort drills.	1	Gen	24 hr/day	1440/day	8440 (1688/day)	Complete command control complex that can be manned continuously by either one or two men. Deep space communication and data transmission systems. Emergency alarm systems and facility for rapid and effective mission abort.	Mission module (MM)	
	b. Navigation and Attitude Correction	1. Take periodic readings of the space vehicle's spatial position. 2. Correlate navigational data with Earth-based tracking data. 3. Update computers and determine required midcourse correction. 4. Assure proper vehicle attitudes during all phases of operation, make necessary corrections, and update computers.	1	N&G and Gen	2 hr/day	120/day		Navigational equipment with manual operation backup. Space vehicle attitude stabilization and control system.		One of the most important tasks to be performed during the first interplanetary coast periods is a navigational task: that of determining the space vehicle's position and trajectory in conjunction with Earth-based tracking stations.
	c. General Housekeeping	1. Recovery of dirt and debris from MM atmosphere and cabin surfaces (vacuuming). 2. Washing and sterilization of equipment as necessary. 3. Sanitized dry cleaning and laundry of linens and clothes.	3	Gen	5 hr/wk	900/wk (128/day)		Vacuum cleaning and filtering system, washing and sterilization provisions and laundry equipment compatible with zero-g or low-g environment	Throughout spacecraft	
	2. Experiment Operations	1. Remove experiment hardware from stowage, checkout and calibrate. 2. Install and/or deploy experiment hardware. 3. Checkout experiment systems and data acquisition, transmission, display, reduction, and storage systems. 4. Adjust space vehicle orientation, as required. 5. Initiate experiment sequence. 6. Evaluate experiment performance, verify completion, and terminate operations. 7. Recover, check out, refurbish and restow experiment hardware as required.	Defined in experiment analysis					Experiment equipment as described in experiments analysis.	Primarily in and around the mission module (MM).	

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4A

147

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147 & 148

FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
[2] (Continued)	3. Space Vehicle Maintenance Operations a. Scheduled Maintenance	1. Routine inspection, servicing, and preventive maintenance operations (e.g., servicing batteries; replenishing cryogenics, gases, water, cooling fluids, propellants; replacement of filters, chemicals, wicking, etc.). 2. Replacement of components due to normal wearout or scheduled replacement (e.g., batteries, reaction control thrusters, etc.).	Defined in maintenance analysis					Maintenance tools (see maintenance analysis), technical maintenance data file, a small workshop area, adjustment and calibration equipment, special handling equipment and storage areas for spare parts and materials, some special shielding and remote handing equipment may be required for emergency work around hot reactors. Transfer Equipment for crew, materials and equipment transfer must be provided including airlocks, pressurized passage ways, hand and foot holds, conveyors, tethers, etc.	Throughout space vehicle	Maintenance tools, equipment, and storage provisions must be compatible with the anticipated zero-g or low-g operation of the space vehicle. Note: Personnel operations, including personal hygiene, crew training, air conditioning, nutrition, recreation, sleep, etc., that must take place during all phases of the mission are shown typically on sheets 16 and 17.
	b. Unscheduled Maintenance	1. Replacement of components because of random failures. 2. Recalibration or adjustments required to bring component operation back within required tolerance. 3. Repair of damage resulting from micrometeoroid impacts, unanticipated human errors during vehicle operation or maintenance or from equipment handling accidents. 4. Replacement of components found defective during scheduled maintenance.								
	Note: (see remarks)									
[3] First Midcourse Correction (0.7 hour)	1. Space Vehicle Orientation	1. Make final determination (prior to midcourse correction) of vehicle's position in space, the trajectory being flown, and the correction required to place the vehicle on the desired trajectory, using on-board navigation systems. 2. Coordinate the above information with Earth-based tracking stations and update computers with the correction data. 3. Orient vehicle to the prescribed correction attitude.	2	FO and N&G	30 min	60	80	Same navigation and guidance system prescribed in [1] and [2] above.	Mission module (MM)	
	2. Correction Maneuver	1. Initiate midcourse correction sequence and monitor its performance. 2. Assure satisfactory completion of the maneuver and make any manual adjustments necessary. 3. Coordinate new flight data with Earth-based tracking.	2	FO and Gen	10 min	20		Midcourse correction module to perform at least three corrections and possibly accomplish planet orbit corrections.	Mission module (MM) and abort stations.	Nonparticipating crew members stand by at abort stations Approximately 6-minute burn time.
[4] Second Interplanetary Coast and Experiments (20 days)	1. Space Vehicle Operations a. Systems Monitoring and Control	Same as for [2].	1	Gen	24 hr/day	1440/day	31,540 (1577/day)	Same as for [2].	Same as for [2].	
	b. Navigation and Attitude Correction	Same as for [2] except navigation tasks not quite as critical.	1	N&G and Gen	1 hr/wk	60/wk (9/day)				
	c. General Housekeeping	Same as for [2].	3	Gen	5 hr/wk	900/wk (128/day)				
	2. Experiment Operations	Same as for [2] except perhaps more extensive.	Defined in experiment analysis							Probably more experiment time.
	3. Space Vehicle Maintenance Operations	Same as for [2].	Defined in maintenance analysis							

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4:A
149

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FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
[5] Second Midcourse Correction (0.6 hour)	1. Space Vehicle Orientation 2. Correction Maneuver	Same as for [3]. Same as for [3] except the actual maneuver may be shorter in duration.	2	FO and N&G	30 min 5 min	70	70	Same as for [3].	Same as for [3].	Correction maneuver will probably be needed for a refinement of the trajectory. only. Approximately 3-minute burn time.
[6] Third Interplanetary Coast and Experiments (245 days)	1. Space Vehicle Operations a. Systems Monitoring and Control b. Navigation and Attitude Correction c. General House-keeping 2. Experiment Operations 3. Space Vehicle Maintenance Operations	Same as for [2]. Same as for [2] except navigational task becomes more critical nearing the 3rd midcourse correction Same as for [2]. Same as for [2]. Same as for [2].	1 1 3 Defined in experiment analysis Defined in maintenance analysis	Gen N&G and Gen Gen	24 hr/day 1 hr/wk 5 hr/wk	1440/day 60/wk (9/day) 900/wk (128/day)	386,365 (1577/day)	Same as for [2].	Same as for [2].	Navigation must prepare for the final outbound midcourse correction and the alignment required for the planet capture maneuver.
[7] Third Midcourse Correction (0.6 hour)	1. Space Vehicle Orientation 2. Correction Maneuver	Same as for [3]. Same as for [3] except of shorter duration burn.	2	FO and N&G	30 min 5 min	70	70	Same as for [3].	Same as for [3].	Correction maneuver will be final refined alignment. Approximately 1-minute burn time.
[8] Fourth Interplanetary Coast and Experiments (20 days)	1. Space Vehicle Operations a. Systems Monitoring and Control b. Navigation and Attitude Correction. c. General House-keeping. d. PM-2 Checkout 2. Experiment Operations 3. Space Vehicle Maintenance Operations.	Same as for [2]. Same as for [2] except more precise position determination with respect to the target planet will be necessary Same as for [2]. 1. Perform electronic control circuitry checkout for PM-2. 2. Perform readiness check of propellant storage and feed systems of PM-2. 3. Perform low power checkout of nuclear reactor of PM-2. Same as for [2] except during the latter part of this coast phase the experiments may have to be performed on a time-available basis. Same as for [2].	1 1 3 2 Defined in experiment analysis Defined in maintenance analysis	Gen N&G and Gen Gen Prop and Gen	24 hr/day 1 hr/day 5 hr/wk 1 hr	1440/day 60/day 900/wk (128/day) 120	32,560 (1628/day) 120	Same as for [2]. Ranging radar may come into use in the latter part of this coast period. Electronic-control systems with builtin trouble shooting feed back circuitry and information display for PM-2. Experiment equipment as described in experiment analysis.	Same as for [2].	PM-2 should be checked out late in this coast period but early enough that corrective action or abort action can be initiated if malfunction possibilities appear.
© Dispose Meteoroid Shield (MS) of PM-2 and Outbound Mid-course Correction Propulsion Module (OBMC-PM) (0.5 hour)	1. Attachment Release 2. Separation of MS	1. Activate release mechanisms for PM-2 meteoroid shield and OBMC-PM. 2. Assure proper mechanisms release. 1. Initiate separation sequence and take corrective action if separation malfunction occurs.	2	FO and Gen	30 min	60	60	Release mechanisms and separation systems for MS of PM-2 with manual release and separation provisions.	Mission module (MM)	In case of a release and/or separation malfunction EVA may be required as corrective action. Provisions for such should be made.

Figure 3.1-4: 6 TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4A

FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
[9] Planet Capture and Orbit Insertion (PM-2)	1. Final Capture Preparation	1. Make precise position determination and correct within capabilities of midcourse PM if required. 2. Determine capture and injection requirements (ΔV, burn time, time for stage ignition, attitude position, etc.) and feed data into computer. 3. Orient vehicle to prescribed maneuver attitude.	2	FO and Gen	2 hr	240		Ranging radar	Mission module (MM)	
(4.5 hours)	2. Engine Ignition.	1. Initiate manual engine startup sequence. 2. Monitor instrumentation and verify satisfactory reactor and propellant systems control.	1	Prop	5 min	5		Same engine and propellant systems and control as required for PM-1 and listed for [1].	Mission module (MM) and abort stations.	
	3. Perform Propulsive Maneuver.	1. Initiate programmed power buildup sequence. 2. Assure satisfactory reactor and propellant systems operation. 3. Assure prescribed vehicle attitude during thrusting. 4. Assure achievement of required incremental velocity change and prepare for engine shut down.	2	FO and Prop	20 min	40	425			
	4. Shut Down Engine	1. Initiate programmed engine shut down sequence. 2. Assure satisfactory power decay and aftercooling. 3. Adjust engine and propellant control operations to the intended stage disposition requirements.			2 hr	240				The amount of aftercooling propellant required will depend on the intended disposition of the stage.
Ⓓ Dispose Spent Stage PM-2 (1 hour)	1. Release Stage Attachment. 2. Separate PM-2.	1. Activate release mechanisms. 2. Assure complete release. 1. Assure PM-2 stage is in separation mode of operation. 2. Separate PM-2 to safe distance from space vehicle. 3. Ignite maneuver stage on PM-2 or restart PM-2 engines and propell stage to desired disposal conditions.	3	FO, Prop and Gen	1 hr	180	180	PM-2 will require an IU and either a separate maneuvering stage or restart capability in the PM-2 engines for boasting the spent stage to high orbit or to escape (see note in remarks). It may be desirable to maneuver the space vehicle away from PM-2, in which case, added capability (propellant) will have to be provided in the orbit trim PM.	Mission module (MM) and abort stations.	Note: Disposal of PM-2 after it has been taken into planet orbit provides interesting problems from the stand points of vehicle design and operation. It may be desirable to retain the PM-2 engines (as prescribed by TRW) and use them for PM-3, in which case, cool down to equilibrium conditions will be necessary. This will require cooling perhaps as long as several days and perhaps a significant amount of added propellant. It will then be necessary to either provide added shadow shielding for the hot engines or restrict all orbital operation around the space vehicle to within the concical area shielded by the integral engine shield. Scattering of fission products into the planet atmosphere may also be a problem.

Fig. 3.1-4: A

Figure 3.1-4: B TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

153 & 154

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FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
[10] Planet Orbit Coast and Experiments (5 days)	1. Space Vehicle Operations									
	a. Systems Monitoring and Control	Same as for [2] .	1	Gen	24 hr/day	1440/day		Same as for [2] .	Same as for [2] .	
	b. Navigation and Attitude Correction	1. Take periodic measurements of the space vehicle's position relative to the planet. 2. Coordinate planet orbit data and space vehicle positions with Earth-based mission control. 3. Update computer and determine the required orbit correction. 4. Provide proper vehicle orientation during navigational sightings and experiments.	2	N&G and Gen	2 hr/day	240/day	9040 (1808/day)	Same as for [2] with whatever additional sensors required for near planet navigation.	Same as for [2] .	Frequent position sightings will be necessary soon after orbit insertion to make an accurate determination of the orbit achieved and the necessary correction to go into the desired orbit.
	c. General Housekeeping	Same as for [2] .	3	Gen	5 hr/wk	900/wk (128/day)				
	d. Orbit Trim Propulsion System Check Out	1. Perform electronic control circuitry checkout. 2. Check propellant storage and feed systems. 3. Check TVC system.	2	FO and Prop	1 hr	120	120			
	2. Experiment Operations	Same general procedures described for [2] except these tasks will involve the intended planet orbital experiments.		Defined in experiment analysis				Planet orbit experiment equipment as described in experiment analysis.		During this first planet orbit coast, little time may be available for experimentation except as may be required to verify environmental parameters that may affect the vehicle flight or performance.
	3. Space Vehicle Maintenance Operations	Same as for [2] .		Defined in maintenance analysis						
[11] Planet Orbit Correction (0.8 hour)	1. Space Vehicle Orientation	1. Make final determination of vehicle's orbital position, the desired orbit entry points, incremental velocity change required, thrust orientation, time and position, and update computer. 2. Orient space vehicle to the prescribed attitude at the required orbital position for ignition.	3	FO, N&G and Prop	30 min 20 min	90 60	150	Same as for [3] and [10] above.	Mission module (MM) and abort stations	
	2. Correction Maneuver	1. Initiate orbit maneuver sequence and monitor systems performance. 2. Make manual adjustments as required and verify satisfactory completion of the maneuver. 3. Determine new orbit data and coordinate data with Earth-based mission control.						Correction may be accomplished by mid-course correction PM or special orbit trim system.		

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4A

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~~155~~ & 156

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FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
Orbit Operations 12a 13a and 14a Coast in Planet Orbit and MEM Monitoring (8.5 hours)	1. Space Vehicle Operations a. Systems Monitoring and Control	Same as for [2].	2	FO and Gen	8.5 hr	1020	1260	Same as for [2]. Space vehicle/MEM communications and data exchange links.	Mission module (MM) and throughout spacecraft	
	b. MEM Operations Monitoring	1. Monitor complete orbital checkout of MEM and MEM separation. 2. Provide orbital mission control for MEM during deorbiting, descent, terminal maneuvers and landing.								
	c. Navigation and Attitude Correction	1. Take periodic measurements of space vehicle's orbital position. 2. Coordinate orbit and vehicle data with Earth-based mission control and update computers.	1	N&G	4.0 hr	240		Same as for [10].		Maintaining correct space vehicle/MEM orientation becomes an extremely critical task in the event PM-2 (with a radioactively "hot" engine) is retained with the vehicle in orbit. This becomes even more critical if no extra shadow shield is provided for the "hot" PM-2.
	2. Experiment Operations	1. These may be ommitted during this period if other activity is too demanding; otherwise same as [10].		Defined in experiments analysis						
	3. Space Vehicle Maintenance Operations	Same as for [2].		Defined in maintenance analysis						
15a Coast in Planet Orbit, Experiments, and Surface Activities Monitoring (29.5 days)	1. Space Vehicle Operations a. Systems Monitoring and Control	Same as for [2].	1	Gen	24 hr/day	1440/day	46,787 (1586/day)	Same as for [2]. Same as for [10]. Same as for [2]. Same as for [10]. Same as for [2]. Same communication system as [12a] above with adequate orbit-to-surface range.	Mission module (MM)	
	b. Navigation and Attitude Correction	Same as for [12a] above except orientation will be maintained to support orbital experimentation and planet surface operations.	1	N&G	1 hr/day	60/day				
	c. General Housekeeping	Same as for [2].	2	Gen	5 hr/wk	600/wk (86/day)				
	2. Experiment Operations	Same as for [10].		Defined in experiments analysis						
	3. Space Vehicle Maintenance Operations	Same as for [2].		Defined in maintenance analysis						
	4. Monitor Planet Surface Exploration Activities	1. Communicate each orbit with surface exploration station (MEM). 2. Relay information as required from MEM to Earth-based mission control.	1	Gen	6 hr/day minimum	360/day	10,620			It may be advisable from a communications standpoint to either adjust the orbit with apoapsis above the surface operations and a highly elliptical orbit to increase the communication time for each orbit or provide three relay satellites in a synchronous orbit to provide continuous orbit-to-surface communications.
16a Orbit Adjustment and MEM Monitoring (3.8 hours)	1. Space Vehicle Orientation	1. Determine vehicle's orbit and the vehicle's position in the orbit with respect to the new orbit or orbit position desired. 2. Determine the correction required and orient space vehicle to the prescribed attitude for the adjustment maneuver.	3	FO, N&G and Prop	30 min	90	690	Same as for [3] and [10] above.	Mission module (MM) and abort stations if maneuver is appreciable.	
	2. Adjustment Maneuver	1. Initiate adjustment sequence. 2. Verify satisfactory completion of maneuver. 3. Determine new orbit and/or position parameters and coordinate with Earth-based mission control.			20 min	60		Same as for [11] above.		
	3. MEM Monitoring	1. Monitor surface activities in readying MEM for return.			3 hr	540				

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4A
157
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FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
17a Coast in Orbit and MEM Monitoring (3.2 hours)	1. Space Vehicle Operations a. Systems Monitoring and Control	Same as for 2.	1	Gen	24 hr/day	1440/day	211 (1586/day)	Same as for 2 and 16a if required.	Mission module (MM)	
	b. Navigation and Attitude Correction	Same as for 12a except an orbit positioning maneuver may be required in preparation for rendezvous with the MEM.	1	N&G	1 hr/day	60/day				
	c. General Housekeeping	Same as for 2.	2	Gen	5 hr/wk	600/wk (86/day)				
	2. Monitoring of MEM in its prelaunch, launch and ascent phases of its return to orbit.	1. Monitor surface prelaunch checkout of MEM. 2. Provide orbital mission control for directing MEM countdown, launch and ascent to orbit. 3. Relay information to Earth-based mission control as required.	3	FO and Gen	3.2 hr	576	576	Same as for 12a.		
	3. Space Vehicle Maintenance Operations		Defined in maintenance analysis							
Surface Exploration Operations 12 MEM Preparation and Deorbit (13 hours)	1. Checkout and Preparation for Separation	1. Checkout MEM systems that can be checked out in the stowed position. 2. Remove from storage area, MEM crew put on pressure suits and transfer to MEM. 3. Complete MEM systems checkout.	3	FO and Gen	5 hr	900	1260	Complete checkout instrumentation on-board MEM and remote checkout instruments on space vehicles control console in MM.	Mission module (MEM) and MEM storage area of spacecraft.	
	2. MEM Separation and Deorbit	1. Coordinate navigational and deorbiting flight data with orbital mission control. 2. Update on-board computers and separate from space vehicle. 3. Position MEM in prescribed attitude and position. 4. Countdown for deorbiting and initiate deorbit sequence. 5. Verify proper retro-operation and proper engine cutoff.	2	FO and N&G	3 hr	360		Equipment capable of moving the MEM from its stored position to docking position. MEM attitude stabilization and control. Retro-propulsion for MEM could be accomplished by the terminal maneuvering and landing engine.	Mars entry module (MEM)	
13 Entry and Descent Operations (0.2 hour)	1. Entry and Descent	1. Position MEM attitude for aero breaking entry. 2. Monitor systems operation during entry. 3. Adjust velocity profile as necessary.	2	FO and N&G	10 min	20	20	Aeroshell to facilitate the maximum expected entry heating conditions. Sufficient entry propulsion to supplement the aero-braking change in velocity.	MEM	

Fig. 3.1-4A

Figure 3.1-4: B TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
14 MEM Terminal Maneuvers and Landing. (0.3 hour)	1. Range Adjustment	1. Determine landing position for the achieved entry trajectory and compare with desired landing site. 2. Determine terminal maneuvers required to achieve desired landing velocities and site. 3. Initiate maneuver sequences and verify correction performance.	2	FO and N&G	20 min	40	40	Same as for 12 .	MEM	
	2. Landing and Maneuvers	1. Scan desired landing area and select touchdown site. 2. Maneuver to final touchdown site and make radar and visual observations of the area. 3. Land MEM at touchdown site. 4. Acquire an accurate fix on landing site and notify orbit control.						Landing radar and visual observation provisions (TV, windows, etc.) Same engine specified in 12 but must accomodate hovering and final touchdown maneuvers (throttleable and restartable).		
15 Planet Surface Exploration (29.5 days)	1. Debarking Preparations	1. Conduct postlanding checkout of MEM. 2. Perform remotely controlled investigation of planet surface and environment immediately around the MEM. 3. Perform final checkout of experiment and personnel equipment.	3	FO and Gen	4 hr	720	720	Same as for 12 . Environment sampling equipment for analysis of atmosphere, radiation flux, surface composition and hardness, etc.	MEM	
		4. Don hard suits, check out and activate exit mechanisms.						Pressurized hard suits with life support backpacks.	MEM and planet surface	Pressurized soft suits should be worn through the entire entry and landing but the hard suits must be worn on the planet surface until all environmental factors have been investigated.
	2. Debarking and Planet Surface Exploration.	1. Partial crew debarks leaving the surface command station of the MEM manned. 2. Exploration crew performs scheduled exploration and experiments, makes observations and performs other pertinent unscheduled tests as determined useful.	Defined in experiment analysis					Exit mechanisms for MEM and provisions for unloading and reloading equipment. Surface experiment equipment as described in Volume III, Part 2.		
	3. MEM Surface Operations	1. Maintain continuous radio contact with the exploration crew. 2. Relay and coordinate pertinent information with orbiting space vehicle. 3. Perform necessary MEM housekeeping and maintenance.	1	Gen	12 hr/day	720/day	21,240	MEM maintenance tool and equipment		
E Dispose Deorbit and Descent PM and Surface Equipment (3 hours)	1. Return to MEM and make initial preparation for launch.	1. Crew returns to MEM, loads required equipment, and samples and discards unnecessary equipment. 2. Release descent stage attachments.	3	Gen	3 hr	540	540	Some type of isolated reentry way to the MEM and sample storage may be required to assure no planet contamination of Earth.	Planet surface and MEM	Descent stage will probably be used as the surface launch pad for the MEM ascent vehicle.

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4A

FUNCTIONS -MAJOR EVENTS	TASKS: DESCRIPTION	TASKS: PROCEDURE	PERSONNEL: NUMBER OF PERSONS	PERSONNEL: SKILLS REQUIRED	TIME: ELAPSED	TIME: MAN-MINUTES PER TASK	TIME: ACCUM MAN-MINUTES	EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
16 MEM Preparation and Launch from Planet Surface (3 hours)	1. Final Launch Preparations	1. Perform prelaunch visual checkout of MEM. 2. Perform prelaunch systems checkout. 3. Coordinate launch window ascent trajectory parameters, etc. with orbiting vehicle.	3	FO, N&G and Gen	2 hr	360	540	Same as for 12.	MEM	
	2. Countdown and Launch	1. Perform coordinated countdown with orbiting mission control (space vehicle). 2. Verify positive release from descent stage. 3. Initiate surface launch sequence.			1 hr	180				
17 MEM Ascent to Orbit (3.1 hours)	1. MEM Ascent	1. Verify achievement of the required velocity and trajectory in coordination with orbital mission control. 2. Make corrections, as required.	2	FO and N&G	5 min	10	370	Ascent stage capable of performing planet launch ascent corrections and orbit insertion (throttle-able and restartable).	MEM	
	2. Parking Orbit Insertion	1. Initiate orbit insertion sequence and verify performance.								
	3. Preparation for Rendezvous Operations	1. Determine parking orbit parameters and position of MEM in the orbit. 2. Coordinate with space vehicle and determine rendezvous maneuver requirements. 3. Initiate ascent maneuver and insertion into space vehicle orbit near space vehicle.	2	FO and N&G	3 hr	360		Navigational systems in MEM capable of handling the surface-to-orbit & orbit change maneuvers required for rendezvous operations.	MEM	Ascent to rendezvous orbit still uses ascent engine.
F Dispose MEM Planet Launch PM (Ascent Stage) (0.1 hour)	1. Release Stage Attachment	1. Activate stage release mechanisms. 2. Assure satisfactory stage release.	2	FO and Gen	5 min	10	10	Positive stage release mechanism	MEM	
End of Surface Exploration Operations	2. Separate Ascent Stage	1. Verify complete deactivation of the stage 2. Activate separation systems. 3. Assure satisfactory separation and relative orientation of space vehicle and ascent stage.						Separation system that can provide the desired separational distance & direction.		
18 MEM and Space Vehicle Rendezvous and Docking and Coast (4.7 days)	1. Rendezvous Positioning	1. Determine separation distance and closing rate between space vehicle and MEM after arrival in close proximity of space vehicle. 2. Adjust closing rate and MEM attitude.	2	FO and N&G	20 min	40	800	Attitude control system can probably be used for fine adjustments & docking maneuvers. Docking system and crew transfer system.	Space vehicle & MEM	Note: If a "hot" PM-2 is kept with the space vehicle, either a rather large radiation shadow shield must be provided or the rendezvous & docking of the MEM must be restricted to the limited shadow cone on the engine's integral shield.
	2. Docking	1. Maneuver MEM into docking mechanism of space vehicle and assure satisfactory attachment.	2	FO and N&G	20 min	40				
	3. Coast and Vehicle Preparation	1. Pressurize transfer systems and transfer samples and equipment to space vehicle. 2. Shut down MEM systems and transfer crew 3. Prepare space vehicle for return trip.	3	Gen	4 hr	720				
	4. Space Vehicle Operations:	1. Same as for 17c except navigation activities will be devoted to return trip preparation.								
	a. System Monitoring and Control		1	Gen	24 hr/day	1440/day				
	b. Navigation and Attitude Correction		2	Gen and N&G	2 hr/day	240/day	8498 (1808/day)			
	c. General Housekeeping		3	Gen	5 hr/wk	128/day				
	5. Experiment Operations	1. Perform final orbital experimentation.	Defined in experiment analysis							

Figure 3.1-4: B TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
(G) Dispose Orbit Trim PM, MEM and Meteoroid Shield of PM-3 (0.7 hour)	1. Release Attachments	1. Activate release mechanisms for MEM, orbit trim propulsion module, and meteoroid shield of PM-3.	2	FO and Gen	30 min	60	100	Release mechanisms (remote controlled) & separation systems for midcourse PM and MS...., of PM-3.		
		2. Assure proper mechanisms release.								
	2. Separate MEM and Orbit Trim PM.	1. Initiate separation sequence and assure proper separation of MEM and orbit trim PM from the space vehicle.								
	3. Separate Meteoroid Shield and Insulation of PM-3	1. Initiate separation sequence and take corrective action if separation malfunction occurs.	2	FO and Gen	20 min	40				
19 Space Vehicle Preparation and Launch from Planet Orbit (PM-3) (3.3 hours)	1. Launch Countdown	Same as for [1].	3	FO, N&G and Gen	1 hr	180	455	Same systems required as for [1].	Mission module (MM) and abort stations	
	2. Engine Ignition	Same as for [1].	1	Prop	5 min	5				
	3. Perform Propulsive Maneuver	Same as for [1].	2	FO and Prop	15 min	30				
	4. Shut Down Engine	Same as for [1].	2	FO and Prop	2 hr	240				
(H) Dispose Spent Stage PM-3 (0.5 hour)	1. Release Stage Attachment	Same as for (B).	3	FO, Prop and Gen	30 min	90	90	Same type of systems for PM-3 disposal as for PM-1 of (B).	Mission module (MM) and abort stations	
	2. Separate PM-3	Same as for (B).								
20 First Return Inter-Planetary Coast and Experiments (5 days)	1. Space Vehicle Operations: a. Systems Monitoring and Control	Same as for [2].	1	Gen	24 hr/day	1440/day	8440 (1688/day)	Same systems as required for [2].	Mission module (MM) and throughout spacecraft	
	b. Navigation and Attitude Correction	Same as for [2].	1	N&G and Gen	2 hr/day	120/day				
	c. General Housekeeping	Same as for [2].	3	Gen	5 hr/wk	128/day				
	2. Experiment Operations	Same as for [2].	Defined in experiment analysis					Experiment equipment as described in experiment analysis.		
	3. Space Vehicle Maint. Ops	Same as for [2].	Defined in maintenance analysis					Same as for [2].		
21 First Return Midcourse Correction (0.6 hour)	1. Space Vehicle Orientation	Same as for [3].	2	FO and N&G	30 min	60	70	Same as for [3].	Mission module (MM) and abort stations	
	2. Correction Maneuver	Same as for [3].			5 min	10				Approximately 1-minute burntime.
22 Second Return Inter-planetary Coast and Experiments (20 days)	1. Space Vehicle Operations: a. Systems Monitoring and Control	Same as for [2].	1	Gen	24 hr/day	1440/day	31,540 (1577/day)	Same as for [2].	Mission module (MM) and throughout spacecraft	
	b. Navigation and Attitude Correction	Same as for [2].	1	N&G and Gen	1 hr/wk	9/day				
	c. General Housekeeping	Same as for [2].	3	Gen	5 hr/wk	128/day				
	2. Experiment Operations	Same as for [4].	Defined in experiment analysis					Experiment equipment as described in experiment analysis.		
	3. Space Vehicle Maintenance Operations	Same as for [2].	Defined in maintenance analysis					Same as for [2].		
23 Second Return Midcourse Correction (0.6 hour)	1. Space Vehicle Orientation	Same as for [5].	2	FO and N&G	30 min	60	70	Same as for [3].	Mission module (MM) and abort stations (if correction is significant)	Refinement Corrective
	2. Correction Maneuver	Same as for [5].			5 min	10		Same as for [3].		Approximately 0.5-minute burntime.
24 Third Return Inter-planetary Coast and Experiments (165 days)	1. Space Vehicle Operations: a. Systems Monitoring and Control	Same as for [2].	1	Gen	24 hr/day	1440/day	260,205 (1577/day)	Same as for [2].	Mission module (MM) and throughout spacecraft	
	b. Navigation and Attitude Correction	Same as for [6].	1	N&G and Gen	1 hr/wk	9/day				Navigation becomes more critical in determining the last midcourse that will align the trajectory for the final entry adjustment.
	c. General Housekeeping	Same as for [2].	3	Gen	5 hr/wk	128/day				
	2. Experiment Operations	Same as for [2].	Defined in experiment analysis							
	3. Space Vehicle Maintenance Operations	Same as for [2].	Defined in maintenance analysis							

Figure 3.1-4:B TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4:A

165

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FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
25 Third Return Midcourse Correction (0.6 hour)	1. Space Vehicle Orientation	Same as for [7].	2	FO and N&G	20 min	60	70	Same as for [3].	Mission module (MM) and abort stations (if correction is significant)	Approximately 0.5-minute burn time.
	2. Correction Maneuver	Same as for [7].			5 min	10				
26 Fourth Return Interplanetary Coast and Experiments (17 days)	1. Space Vehicle Operations									
	a. Systems Monitoring and Control	Same as for [2].	1	Gen	24 hr/day	1440/day	27,676 (1628/day)	Same as for [2].	Mission module (MM)	
	b. Navigation and Attitude Correction	Same as [2] except more precise position determination and trajectory definition will be required to establish inertial reference platform and prepare for Earth entry.	1	N&G and Gen	1 hr/day	60/day		Same as for [8] except Earth-based tracking can provide greater support.		
	c. General Housekeeping	Same as for [2].	3	Gen	5 hr/day	128/day		Same as for [2].		
	d. EEM Entry Guidance System Checkout	Perform electronic control circuitry checkout for entry guidance system.	2	FO N&G and Gen	2 hr	240	240	Electronic circuitry checkout instrumentation		
	e. Recovery System Checkout	Checkout readiness of parachutes and flotation bags.								
	2. Experiment Operations	Same as for [2] except on a time-available basis.		Defined in experiment analysis						
	3. Space Vehicle Maintenance	Same as for [2].		Defined in maintenance analysis					MM and throughtout spacecraft	
	4. Preparation of EEM	1. Activate and checkout EEM systems.	3	FO and Gen	5 hr	900	900	Complete EEM checkout instrumentation	MM and EEM	
		2. Perform integrated systems checkout of EEM vehicle.						EEM maintenance and repair tools and equipment		
		3. Prepare for crew transfer to EEM and separation of EEM from remainder of vehicle.								
① Dispose Inbound Midcourse Correction PM, and Mission Module (MM) (2.3 hours)	1. Release Midcourse PM Attachment and Separate	1. Activate release mechanisms of inbound midcourse propulsion module and verify satisfactory release.	2	FO and Gen	20 min	40	760	Positive release mechanisms for inbound midcourse propulsion module	Mission module (MM)	
		2. Initiate separation sequence and assure proper separation.								
	2. Preparation and Separation of Mission Module (MM)	1. Activate EEM systems and perform final checkout of MM-EEM readiness for separation.	6	FO and Gen	2 hr	720		Same as for [24].	EEM and MM	
		2. Transfer planet exploration samples, data, and equipment to EEM.								
		3. Transfer crew to EEM and shutdown systems of mission module (MM).							EEM	
		4. Activate release mechanism for MM.						Positive release mechanism on MM/EEM attachments.		
		5. Initiate separation sequence and assure proper separation.								

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

167 Fig. 3-1-4:A

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168 & 168

FUNCTIONS -MAJOR EVENTS	TASKS		PERSONNEL		TIME			EQUIPMENT REQUIRED	LOCATION OF TASK	REMARKS
	DESCRIPTION	PROCEDURE	NUMBER OF PERSONS	SKILLS REQUIRED	ELAPSED	MAN-MINUTES PER TASK	ACCUM MAN-MINUTES			
27 Earth Atmosphere Entry (1.6 hours)	1. Preparation for EEM Entry Maneuvers	1. Determine trajectory parameters and coordinate with Earth-based mission control and determine the entry maneuver requirements. 2. Make final check of entry guidance system and orient EEM to correct entry attitude. 3. Make final selection of program inputs for entry maneuvers, and verify with Earth-based mission control.	2	FO and N&G	1 hr	120	200	Approach and entry navigation and guidance systems with Earth/space vehicle communications and data link.	EEM	
	2. Entry Maneuvers	1. Position EEM to the required entry attitude. 2. Monitor entry guidance system and roll angle to achieve constant altitude and required velocity reduction. 3. Initiate skipout maneuver if required.	2	FO and N&G	40	80		Entry systems and biconic aeroshell capable of entry heating commensurate with 65,000-fps entry velocity. Attitude stabilization system for entry conditions.		
	3. Descent Operations	1. Monitor systems operation and make attitude corrections as required. 2. Maintain required glide path.	2	FO and N&G						
28 EEM Terminal Maneuvers, Landing and Recovery Preparation (0.8 hour)	1. Terminal Maneuvers	1. Make attitude corrections as required preparatory to decel chute deployment. 2. Monitor systems operations, deploy chute and coordinate with Earth-based mission control.	2	FO and N&G	15 min	30	210	Deceleration and final descent chutes.	EEM	For emergency situations where survival and recovery of the EEM and crew may be in doubt, it may be advisable to provide an ejectable capsule (containing planet samples and pertinent records) which could survive entry and be fairly easily recovered.
	2. Landing	1. Assure proper EEM attitude for impact. 2. Activate recovery beacon and put EEM systems in impact and recovery mode of operation. 3. Prepare for emergency evacuation from the EEM.						Recovery assist and emergency exit and survival equipment.		
	3. Recovery Preparations	1. Following impact, deactivate EEM systems no longer required and initiate recovery assist operations.	6	Gen	30 min	180				

Figure 3.1-4: TYPICAL MISSIONS OPERATIONS - FUNCTION AND TASK ANALYSIS (Continued)

Fig. 3.1-4: A
169

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~~169~~ & 170

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3.2.1.1 Selected Orbital Support Sequence

A summary chart of the selected sequential flow of the orbital support operations is shown in Figure 3.2-1. This flow chart shows the major orbital support operations required for any of the missions. Orbital operations start with launch (unmanned) of the mission payload, the mission module serving as command center and living quarters for the assembly, test, and checkout (ATC) crew during all subsequent operations. Next, the ATC crew is launched in a logistic vehicle, along with necessary supplies and support equipment. Depending on the particular configuration, additional support equipment may be launched on additional logistic spacecraft. Additional logistic launches and ATC crew rotations will be made as required until completion of the orbital operations. The propulsion modules are launched and assembled in order from PM-3 down through PM-1, using one launch for PM-3, one for PM-2, and three for PM-1. The mission crew is sent up and participates in final checkout, taking over from the ATC crew before their departure. This total operation will require about 140 days.

The major choices made in arriving at the described orbital operations are shown in Table 3.2-1, which lists the selected modes and principle alternates for various mission support categories. In Category 1, the Earth launch of fully fueled modules was selected to simplify orbital operations; i.e., no fuel transfer is required. In Category 2, burdening the mission crew with the additional requirements for assembly operations seems unreasonable, especially considering that the total assembly operation extends over a period of 4 1/2 months. In Category 3, the alternates would require manrating the mission payload/Earch launch vehicle configurations and would impose a length and weight constraint on the design of the standard propulsion module in order to accommodate a logistic spacecraft on top of the stack. It is probable that the additional design and testing costs for manrating would exceed the payoff, whereas a logistic capability must be provided anyway, to satisfy unforeseen requirements for spares, special test equipment, and subsystem specialist crewmen during the orbital assembly and checkout phase.

In Category 4, the alternate space tug mode would require the development of an additional complex space vehicle and therefore was not chosen. For the selected mode, a transtage will be required for each payload that performs a rendezvous maneuver, but it can be standardized and launched with the payload, simplifying the orbital operations relative to the space tug mode. As it builds up, the space vehicle configuration would require excessive propellant for the docking maneuvers and, therefore, the second alternate was not given serious consideration.

3.2.2 EARTH-BASED SUPPORT OPERATIONS

The objectives and scope of this study necessitated only a general understanding of the sequential operations required for establishing general facilities and equipment for costing and schedule estimates for program planning. Figure 3.2-2 represents a typical sequential flow of the ground support operations necessary to receive, inspect, assemble, and test all elements of the space vehicle and the supporting ELV's and to

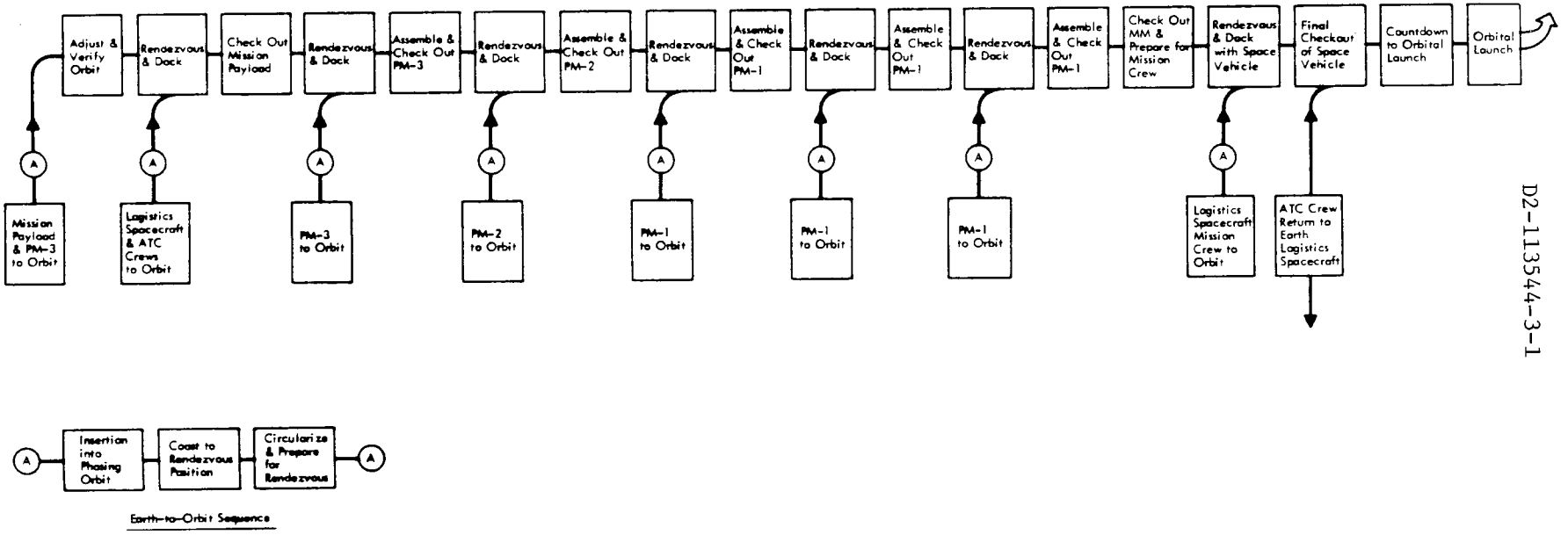


Figure 3.2-1: ORBITAL SUPPORT OPERATIONS SEQUENTIAL FLOW

Table 3.2-1: SELECTED ORBITAL SUPPORT MODES

	Mission Support Mode Category	Selected Modes	Alternate Modes
1.	Earth Launch Payload	Payloads fully fueled	Payloads partially fueled Assembled space vehicle partially fueled Assembled space vehicle fully fueled
2.	Assembly, Test, and Checkout (ATC)	ATC Crew	Mission crew Mission crew + ATC crew
3.	Manning	All men launched by logistic vehicles	All men launched on mission payloads Men launched on logistic vehicles and mission payloads
4.	Rendezvous and Docking	Payload active - Space vehicle passive	Space tug Space vehicle active - Payload passive

integrate these elements into launch packages that will ensure on-time delivery of the space vehicle into the assembly orbit. The ELV, payload, and logistics vehicle sequences are repeated as required to accomplish the required orbital deliveries for a particular mission.

In addition to these ground operations, the mission control, tracking, communications data management, and recovery operations also play an essential part in the total program. For this study, these operations were accounted for only in terms of facilities, equipment, and operational cost and are discussed in Volume IV, Section 3.0.

3.3 CREW OPERATIONS

The objective of this analysis was to estimate crew workloads and assess crew skill requirements for each phase of the mission from Earth orbital launch through Earth reentry and recovery.

The basis for the analysis was the experiment program, the mission vehicle operations, and a maintenance analysis. Experiment programs are presented in Part 2 of Volume III, and a function and task analysis was defined in Section 3.1 of this document. Space vehicle configurations were developed in accordance with the established operational requirements and a maintenance analysis was performed on one configuration. On the basis of these analyses, crew tasks were defined in four categories: vehicle operations, vehicle maintenance, experimentation and exploration, and crew care. Task duration and manloading were estimated and crew skill requirements were identified. Each category is discussed separately, following the summary of the total mission requirements.

3.3.1 WORK LOADS

Crew tasks and skill requirements were investigated in detail using the 1982 Mars opposition mission as a representative case. The total estimated manhours required to accomplish each type of operation during each phase of the mission are shown, along with the average number of men required, in Table 3.3-1. Totals are accumulated for each mission phase. All elements of the planet operations phase are critical for a six-man crew complement. The on-orbit experiment phase requires about 3.2 men on the average to handle experiments and spacecraft operations. During 30 days of the 40-day orbit period, only three men are available on orbit, while the other three have descended to the planet surface. During descent and ascent of the Mars excursion module (MEM), the workload slightly exceeds the capability of the three-man MEM crew and during the 30-day surface phase, the analysis indicates an average requirement of 2.8 men. At the present stage of planning, these results should not be interpreted as proof that a six-man crew is or is not adequate. Further analysis, with a detailed knowledge of the experiment equipment, is needed to ensure the accomplishment of the scientific mission.

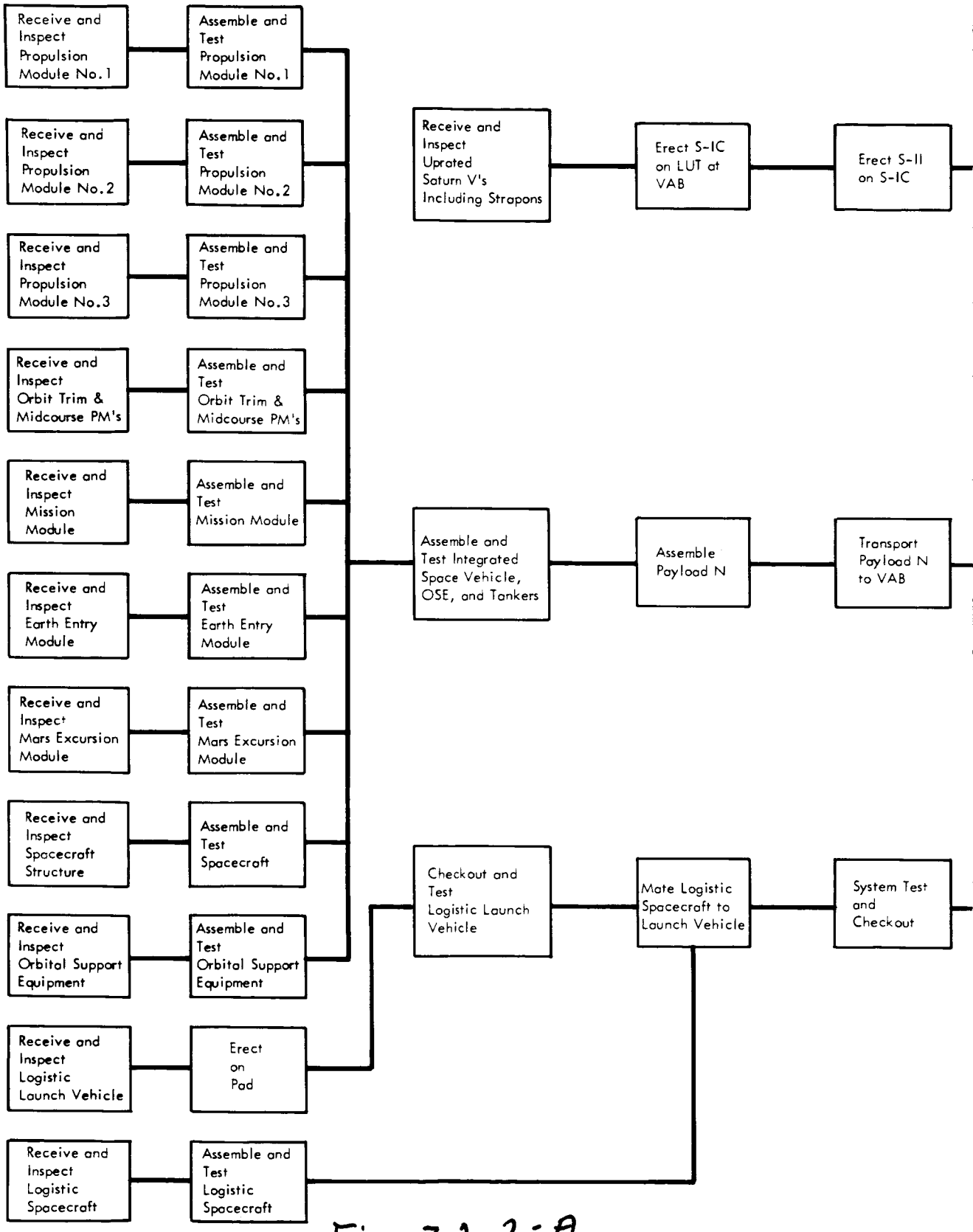
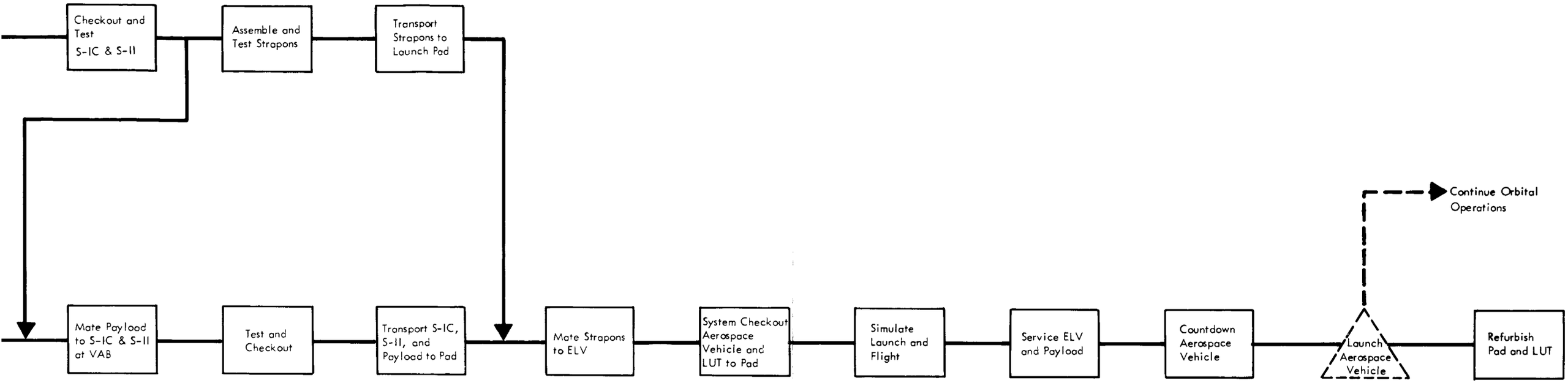

Receive and Inspect Propulsion Module No. 1
Assemble and Test Propulsion Module No. 1
Receive and Inspect Propulsion Module No. 2
Assemble and Test Propulsion Module No. 2
Receive and Inspect Propulsion Module No. 3
Assemble and Test Propulsion Module No. 3
Receive and Inspect Orbit Trim & Midcourse PM's
Assemble and Test Orbit Trim & Midcourse PM's
Receive and Inspect Mission Module
Assemble and Test Mission Module
Receive and Inspect Earth Entry Module
Assemble and Test Earth Entry Module
Receive and Inspect Mars Excursion Module
Assemble and Test Mars Excursion Module
Receive and Inspect Spacecraft Structure
Assemble and Test Spacecraft
Receive and Inspect Orbital Support Equipment
Assemble and Test Orbital Support Equipment
Receive and Inspect Logistic Launch Vehicle
Erect on Pad
Receive and Inspect Logistic Spacecraft
Assemble and Test Logistic Spacecraft
Receive and Inspect Uprated Saturn V's Including Strapons
Erect S-IC on LUT at VAB
Erect S-II on S-IC
Assemble and Test Integrated Space Vehicle, OSE, and Tankers
Assemble Payload N
Transport Payload N to VAB
Checkout and Test Logistic Launch Vehicle
Mate Logistic Spacecraft to Launch Vehicle
System Test and Checkout

Fig. 3.2-2-A



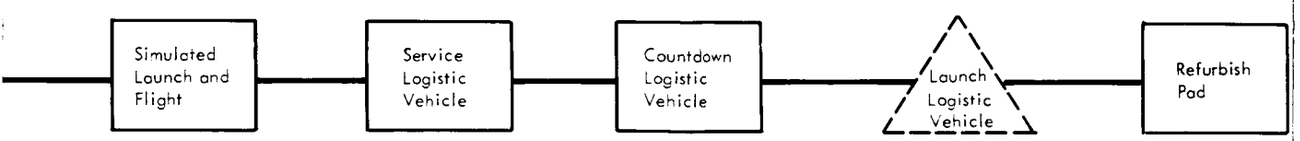


Fig. 3.2-2-B

175B

Figure 3.2-2: C EARTH-BASED SUPPORT SEQUENTIAL FLOW

~~175~~ & 176

Table 3.3-1: NOMINAL CREW TASK REQUIREMENTS SUMMARY -- 1982 MARS OPPOSITION

Mission Phase	Duration	Crew Operations Requirements								Accum. Average No. of Men	Total Manhours
		Vehicle Operations*		Vehicle Maintenance		Expm't & Explor.		Crew Care**			
		Average No. of Men	Total Manhours	Average No. of Men	Total Manhours	Average No. of Men	Total Manhours	Average No. of Men	Total Manhours		
Earth Orbital Launch	5.0 hr	2.52	12.6	Emergency only		—	—	Emergency only		2.52	12.6
In-Transit											
Midcourse Corrections	3.7 hr	1.95	7.2	0.10	1,193	0.34	4,056	3.30	39,369	1.95	7.2
										3.74	44,618
Routine	497 days	1.10	13,132							1.10	13,132
In-Transit Total	497 days	1.10	13,139	0.10	1,193	0.34	4,056	3.30	39,369	4.84	57,757
Planet Capture	6.0 hr	1.85	11.1	Emergency only		—	—	Emergency only		1.85	11.1
Planet Exploration											
Orbital Operations	40.0 days	1.35	1,299	0.10	96	0.47	458	1.25	1,200	3.18	3,048
MEM Descent & Ascent	27.4 hr	2.18	59.7	Emergency only		Emergency only		0.91	25	3.10	85
Surface Operations	30.0 days	0.51	366	0.05	36	1.0	720	1.25	900	2.81	2,022
Planet Exploration Total	40.0 days	1.80	1,725	0.14	132	1.95	1,876	3.28	2,125	5.38	5,155
Planet Orbital Launch	4.5 hr	2.40	10.8	Emergency only		—	—	Emergency only		2.40	10.8
Earth Arrival											
Earth Capture	4.3 hr	3.84	16.5	—	—	—	—	—	—	3.84	16.5
EEM Entry & Landing	0.8 hr	1.88	1.5							1.88	1.5
Earth Arrival Total	5.1 hr	3.53	18.0	Emergency only		—	—	Emergency only		3.53	18.0

*Vehicle operations include staging tasks as applicable in each mission phase.

**Based on a 6-man crew.

3.3.2 CREW SKILLS

Crew skill requirements are summarized in Table 3.3-2 and are discussed under each operational category in the following paragraphs.

3.3.3 VEHICLE OPERATIONS

The vehicle operations category includes flight maneuvers, staging of meteoroid shields and used stages, MEM and EEM operations, and routine vehicle operations during the interplanetary coasting and planet orbiting periods.

Function and task analysis sheets (Figure 3.1-4) with the manpower, time, and skill estimates are included at the end of Section 3.1 of this document.

3.3.3.1 Task Requirements

For summary purposes, the vehicle operation tasks and estimated manhour requirements, as defined in the function and task analysis, were grouped into three categories: flight operations, staging operations, and routine vehicle operations. Flight operations include all major space vehicle maneuvering and final preparations for those maneuvers. The manhours estimated to accomplish the tasks in this category constitute only about 0.7% of the total vehicle operation manhours. Staging operations, which include the tasks of disposing of used propulsion stages, meteoroid shielding, and used MEM and MM, likewise will require a very small percentage of the total vehicle operation manhours (estimated at about 0.2%). The major manhour requirement in vehicle operations fails in the routine vehicle operations category. About 99% of the estimated manhours is required in monitoring and controlling systems operation, navigation and attitude corrections, and in general housekeeping.

Altogether, the required work estimated for performing the vehicle operations for the 1982 Mars opposition, 540-day mission totaled about 14,900 manhours. The distribution of these manhours is illustrated in Figures 3.3-1 and 3.3-2. The average number of crewmen required is also plotted for each general area of tasks. The major manhour requirements in the flight operations stem from the MEM descent and ascent operations which include the MEM's preparation, deorbiting, descent, landing, preparation for return, launch and ascent, rendezvous, and docking. The number of crewmen required is not high because the major portion of the work is MEM preparation stretched out over several hours. The highest crew involvement is imposed by the orbit correction and MEM monitoring tasks where the piloting and monitoring tasks are intensified over the short operational periods. Most of the tasks that involve primarily piloting tasks impose fairly low manhour requirements and a crew utilization of two men on the average. The Earth orbital launch tasks would be expected to be higher than shown if the full countdown

Table 3.3-2: CREW SKILL REQUIREMENTS SUMMARY

Vehicle Operations	Flight Operations (FO)		Propulsion (Prop)
	Navigation and Guidance (N&G)		
Vehicle Maintenance	Life Support (LS)		Communications (Com)
	Electrical/Electronic (E/E)		Mechanical (Me)
	Guidance and Control (G&C)		Structures (S)
	Experimentation (Ex)		
Experimentation and Exploration	Physics (Py)	Optics	Photographic analysis
		IR & UV visible	Mass spectrometry
		Spectral analysis	Nuclear instrumentation
	Astrophysics (A)	Astronomy	Interplanetary physics
		Planetology	Nuclear physics
		Solar physics	Atmospheric physics and meteorology
	Geology (G)	Geology	Physical chemistry
		Seismology	X-Ray defraction
	Communications and Electronics (C&E)	Radar image interpretation	
		Radio wave propagation	
		Electronic equipment	
	Bioscience (B)	Biology	Botany
		Physiology	Radio biology
Crew Care	Medical Treatment		
	Recreation and Psychiatric Service		(MD)
General	Console Monitoring and Operation		Operation of
	First Aid		Life support and ECS,
	EVA		Communication system,
	Mobility in Zero		Electrical power system,
			Emergency abort, Computer systems, Personnel systems and Equipment

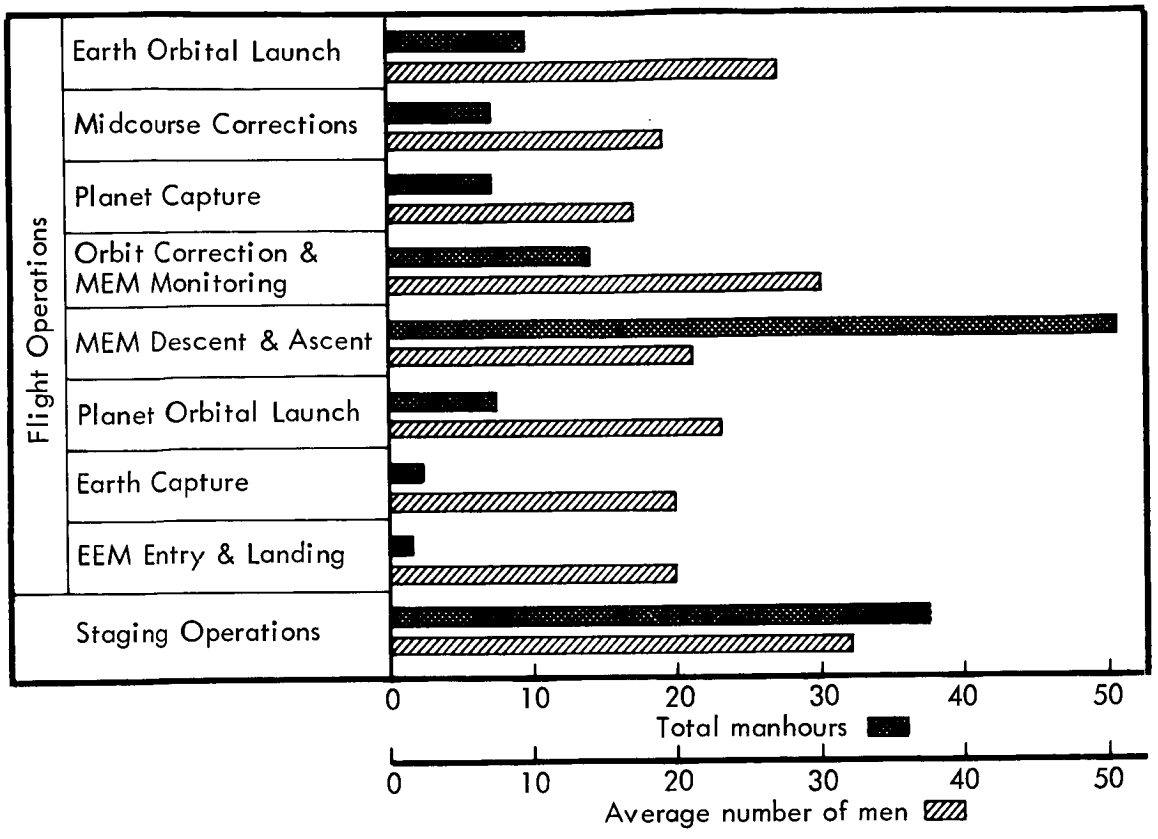


Figure 3.3-1: ESTIMATED WORK REQUIRED IN VEHICLE FLIGHT & STAGING OPERATIONS

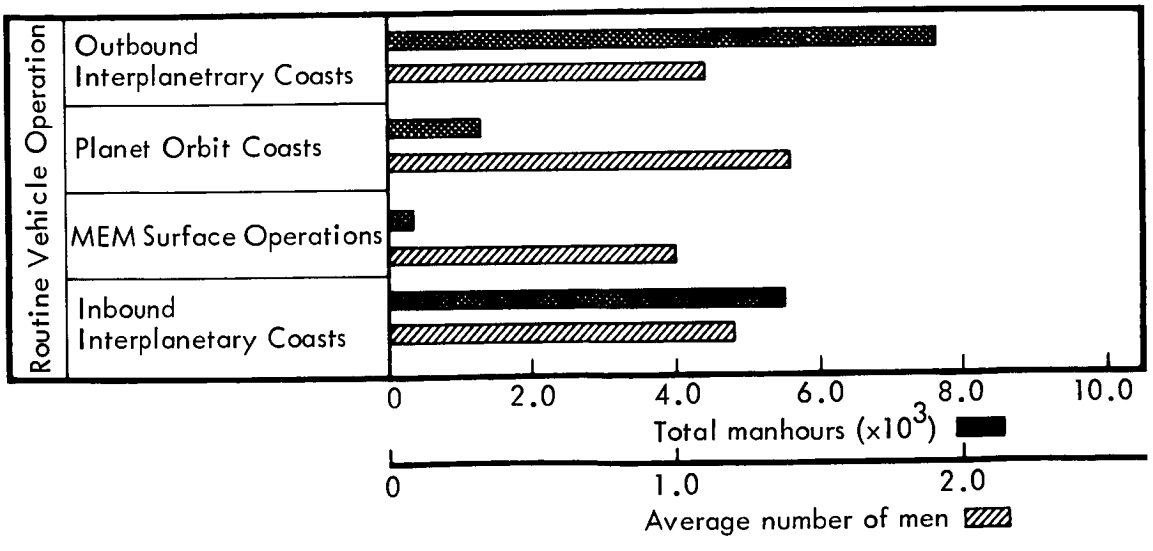


Figure 3.3-2: ESTIMATED WORK REQUIRED IN ROUTINE VEHICLE OPERATIONS

operations were included. The function and task analysis for the mission operations started with the disposal of the meteoroid shield of PM-1 and PM-1 ignition.

The staging operations, also shown in Figure 3.3-1, require an estimated 37.5 manhours of work during the mission, utilizing about three men on the average. These are based on 10 staging operations which necessitate comparatively intense work during the disposal periods.

In Figure 3.3-2, the outbound interplanetary coast area of routine vehicle operation tasks imposes the largest manhour requirement because it spans the greatest time period of the mission (approximately 290 days). The highest work rate is required during the coast in planet orbit where more frequent communications with Earth and MEM are necessary and where knowing and maintaining orbital position necessitates more navigational work. Added MEM and surface operations monitoring contribute to the work load during that period.

3.3.3.2 Skills Requirements

Previous studies, by Boeing and others, have identified a variety of crew skills required to perform manned space missions. All of the studies have concluded that more than one person should be trained in each of the vehicle operation and maintenance skills. This study also assumes that primary and secondary skill training will be necessary to provide backup capabilities in vital areas. The primary skill level is considered "expert" level, where the person is specially trained and highly proficient, capable of making on-the-spot evaluations and changes, if necessary, with little or no outside consultation. Secondary skills are considered "technician" level, where the person is fairly proficient in performing the basic tasks but requires direction or consultation for detailed evaluations or consideration of operational variations.

In vehicle operations, the primary skill requirements can be grouped into flight operations (FO), navigation and guidance (N&G), propulsion (Prop), and general (Gen). Flight operations skills include capabilities for flight command, the capability of interpreting navigation and vehicle performance data, and executing all vehicle flight maneuvers, stage or module separations, rendezvous and docking, entry flight, and landings in MEM or EEM. The manhour requirements estimated for this particular skill only amount to about 0.5% of the total vehicle operation manhours. The navigation and guidance skills for this category chiefly require a high proficiency in celestial navigation and in the operation of all on-board navigational equipment including the emergency manual equipment and on-board computers. The propulsion skills are similar to those required of an aircraft flight engineer. This constitutes complete familiarity with all on-board propulsion systems and high proficiency in initiating, monitoring, and controlling propulsion systems operation. General skills include all of those capabilities required of a crew in maintaining a state of operability in themselves and the space vehicle and in performing the general operations necessary to accomplish the mission. In the vehicle operations category, the

general skills include console operation, operation of communications systems, environmental control systems, electrical power system, emergency abort systems, and computer systems with the capability of assisting in various vehicle operations tasks under the direction of the specialist(s) in the particular skills.

For the 540-day mission, the distribution of the estimated manhour requirements for each skill is as shown in Table 3.3-3. The general skills, primarily involved in performing most of the routine vehicle operations, make up the greatest demand in the vehicle operation skills. Since the FO, N&G, and Prop skills are not only highly critical to successful performance of the mission, but also basic to total crew survival, probably two persons should have primary training in each of these skills, with perhaps one other trained to a secondary skill level. All crew members will be trained in the general skills.

Table 3.3-3: DISTRIBUTION OF VEHICLE OPERATIONS SKILLS REQUIREMENTS

Skill	% of Vehicle Operations Manhour Requirements
Flight Operations (FO)	0.5
Navigation and Guidance (N&G)	0.9
Propulsion (Prop)	0.1
General (Gen)	98.5
	100.0

3.3.4 VEHICLE MAINTENANCE

Maintenance time and skill requirements were obtained from data generated under a Boeing study contract,* "Maintainability of Manned Spacecraft for Long Duration Flights." In that study, a computer program called MARCEP (Maintainability and Reliability Cost Effectiveness Program) was used to optimize the IMISCD spacecraft configuration for least weight, including redundancy and spares, to achieve a probability of mission success of 0.99 for a 1985 Mars mission of 460 days duration. For this optimized configuration, scheduled maintenance requirements were defined, considering routine inspection, servicing and preventive maintenance operations as well as the replacement of components due to normal wearout. Complete maintenance task analysis worksheets are provided in the reference.

Mission simulations, in which random failures are generated and the resulting unscheduled maintenance requirements are calculated, were

*Boeing Document D2-113204, *Maintainability of Manned Spacecraft for Long-Duration Flights*, NASA Contract NAS2-3705, July 1967.

then run. Statistics were generated for a total of 100 mission simulations. The following paragraphs summarize the results as used in the IMISCD study.

3.3.4.1 Task Requirements

An average of 2.1 hours per day were found to be required for regular scheduled maintenance. The mean unscheduled workload determined from the mission simulations and spread over the full 460 days of the mission is only 0.3 hour per day.

Figures 3.3-3, -4, -5, and -6 show plots of some of the results of the unscheduled maintenance simulation program computer runs. It is particularly intersting to note that no unscheduled maintenance can be expected during approximately 92% of the mission and only during 0.2% of the mission will there be unscheduled maintenance that could require an effort of from 20 to 100 manhours. Of the total unscheduled maintenance tasks, 95% will take 10 hours or less to complete, while 30% will require only 30 minutes or less. The mean time per task is about 2.6 hours. About 12 airlock uses per mission can be expected, on the average, for unscheduled maintenance, and the mean space suit usage per mission for unscheduled maintenance could be expected to run about 60 manhours.

3.3.4.2 Skill Requirements

The skills required for vehicle maintenance are listed in Table 3.3-4 along with the subsystem maintenance responsibility assigned to that skill category. The distribution of anticipated scheduled and unscheduled maintenance work by skills is presented in Figure 3.3-7. The cross-hatched portions represent the combination of scheduled and unscheduled skill requirements. This combined plot presents the best estimate of the distribution of maintenance crew skill requirements, since the scheduled maintenance workload is much greater than that of the unscheduled work (by a ratio of 7 to 1).

3.3.5 EXPERIMENTATION AND EXPLORATION

The experimentation proposed for the Mars landing and Venus capture missions is described in Part 2 of Volume III. The experiments are grouped into four categories: in-transit experiments, Mars orbit experiments, Mars surface experiments, and Venus orbit experiments. It was assumed that the in-transit experiments would be generally about the same for Mars or Venus missions, although the equipment may differ somewhat, particularly in operating ranges and sensitivities. Crew operating requirements in terms of number of persons, elapsed time, and skills were defined for the experimentation in each category.

3.3.5.1 Task Requirements

The experiment crew tasks consist of removing the equipment from stowed positions, equipment checkout and calibration, installation of equipment in its operational position, checkout of data display, storage and transmission systems, orientation of the vehicle as necessary, performing

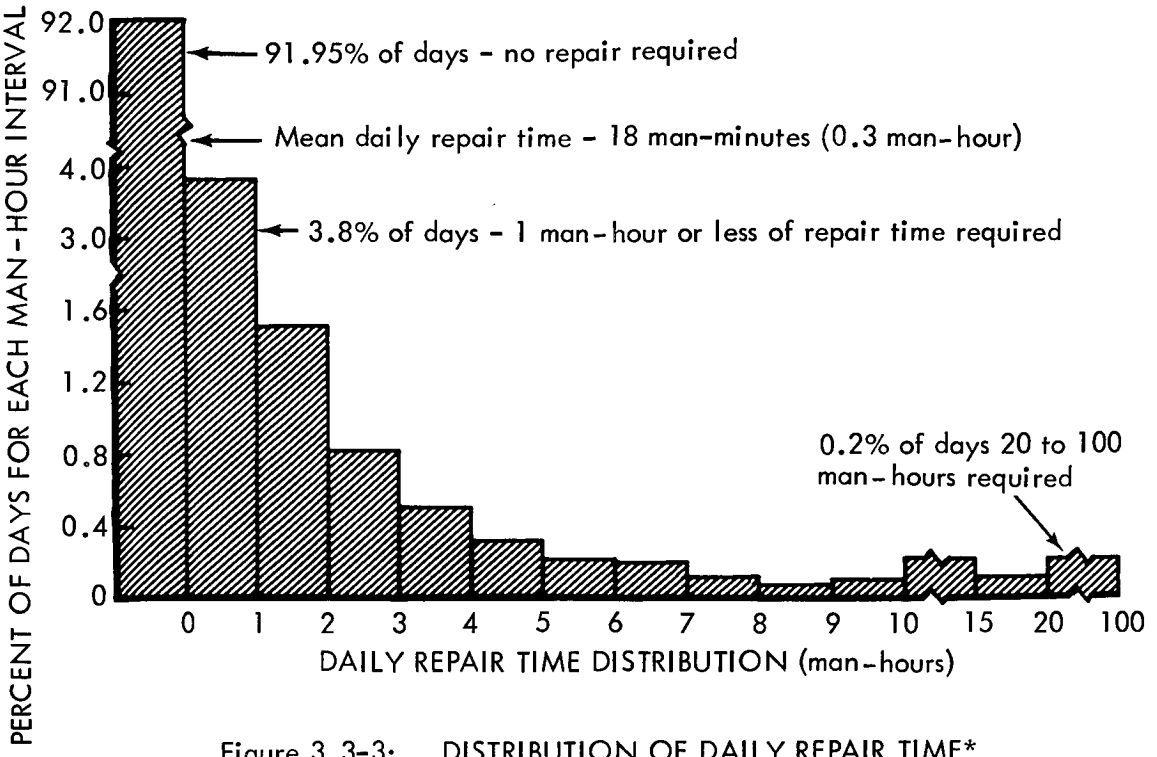


Figure 3.3-3: DISTRIBUTION OF DAILY REPAIR TIME*
(1985 Mars Landing Mission)

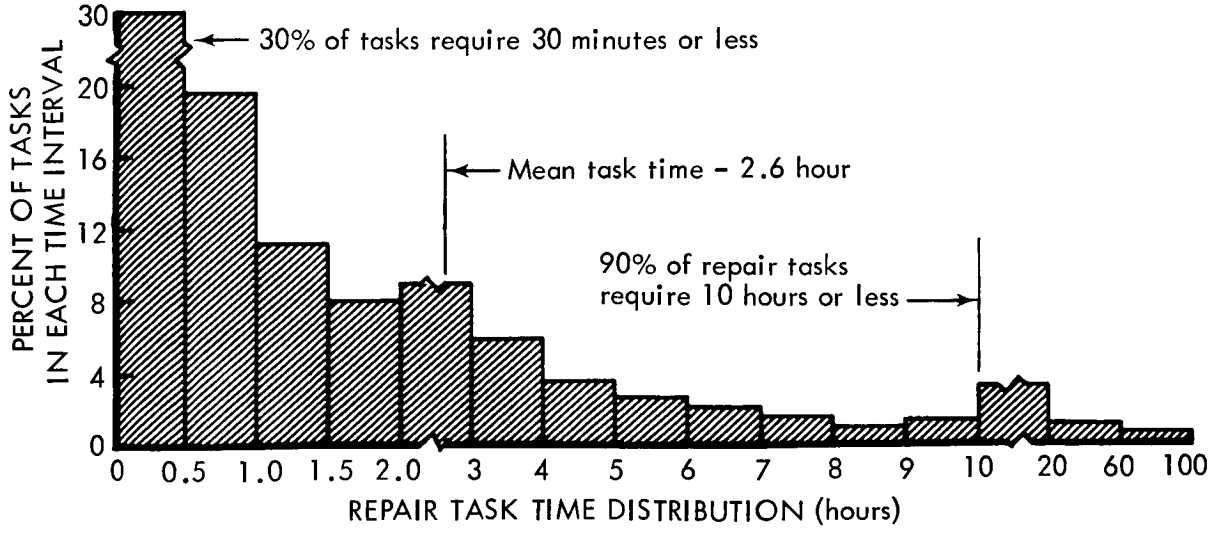


Figure 3.3-4: DISTRIBUTION OF REPAIR TASK TIME*
(1985 Mars Landing Mission)

*Reference: Boeing Document D2-113204-2, Maintainability of Manned Spacecraft for Long Duration Flights, Contract NAS2-3705, July 1967

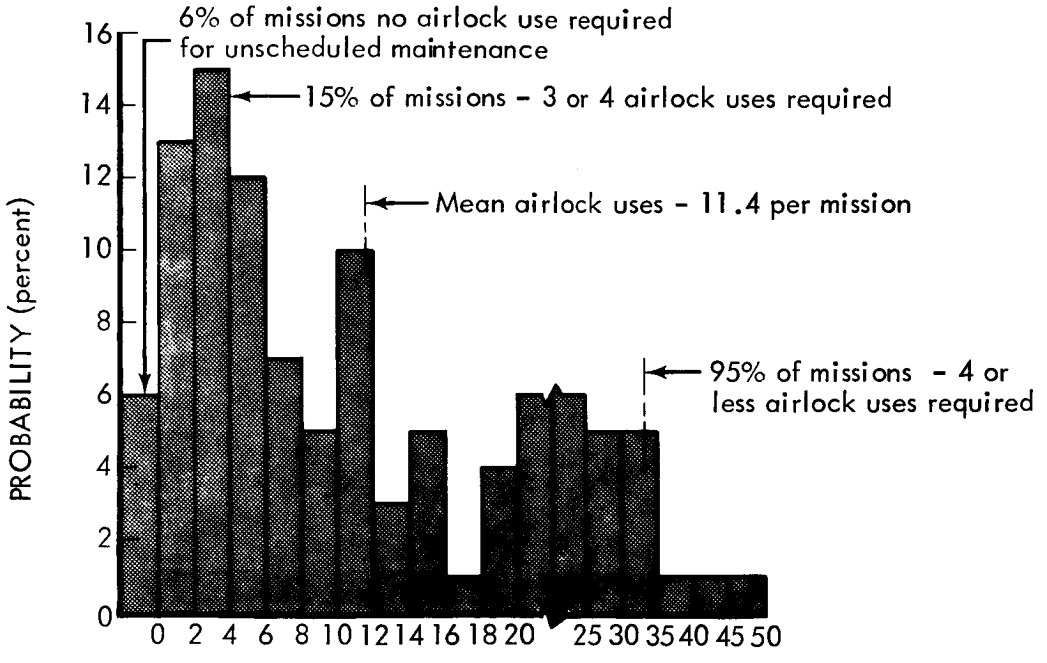


Figure 3.3-5: AIRLOCK USAGE — UNSCHEDULED MAINTENANCE* (1985 Mars Landing Mission)

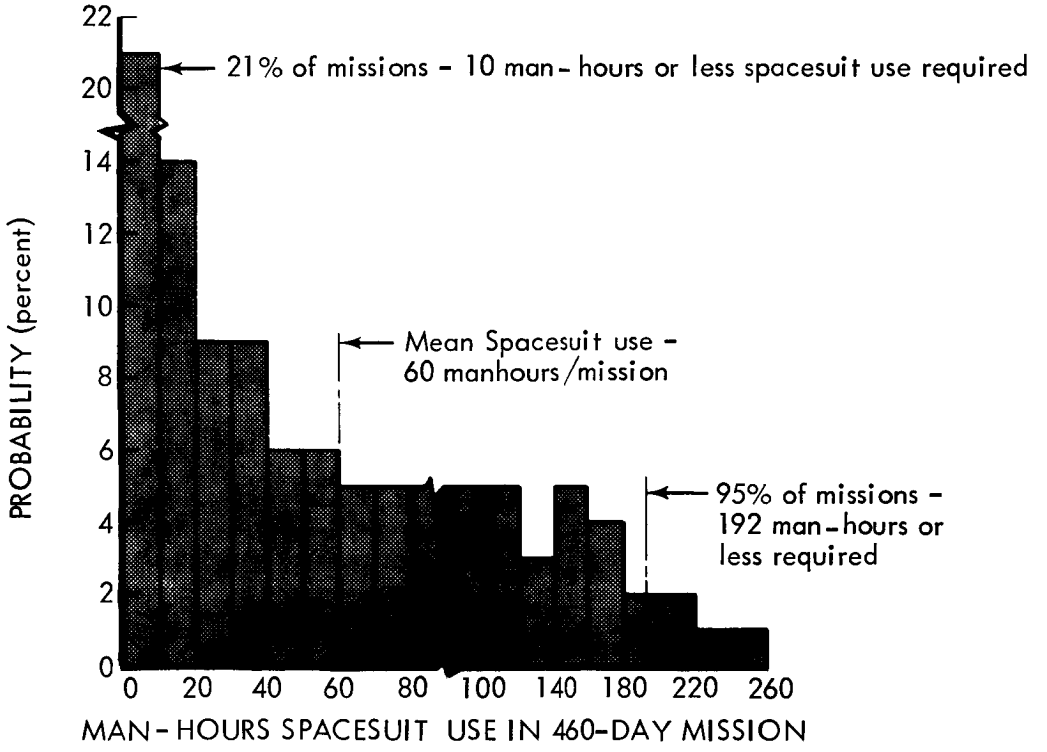


Figure 3.3-6: SPACESUIT USAGE — UNSCHEDULED MAINTENANCE* (1985 Mars Landing Mission)

*Reference: Boeing Document D2-113204-2, Maintainability of Manned Spacecraft for Long Duration Flights, Contract NAS2-3703, July 1967

Table 3.3-4: CREW MAINTENANCE SKILLS/SUBSYSTEM ASSIGNMENTS

Skill	Subsystem Maintenance Responsibility
Life Support (LS)	Environmental control/life support Crew systems
Electrical/Electronic (E/E)	Electrical power Flight electronics
Mechanical (ME)	Mechanical equipment Propulsion Terminal recovery systems
Structures (S)	Structures
Communications (COM)	Communications and data handling
Guidance & Control (G&C)	Guidance and navigation Attitude control
Experimentation (EX)	Experimental equipment

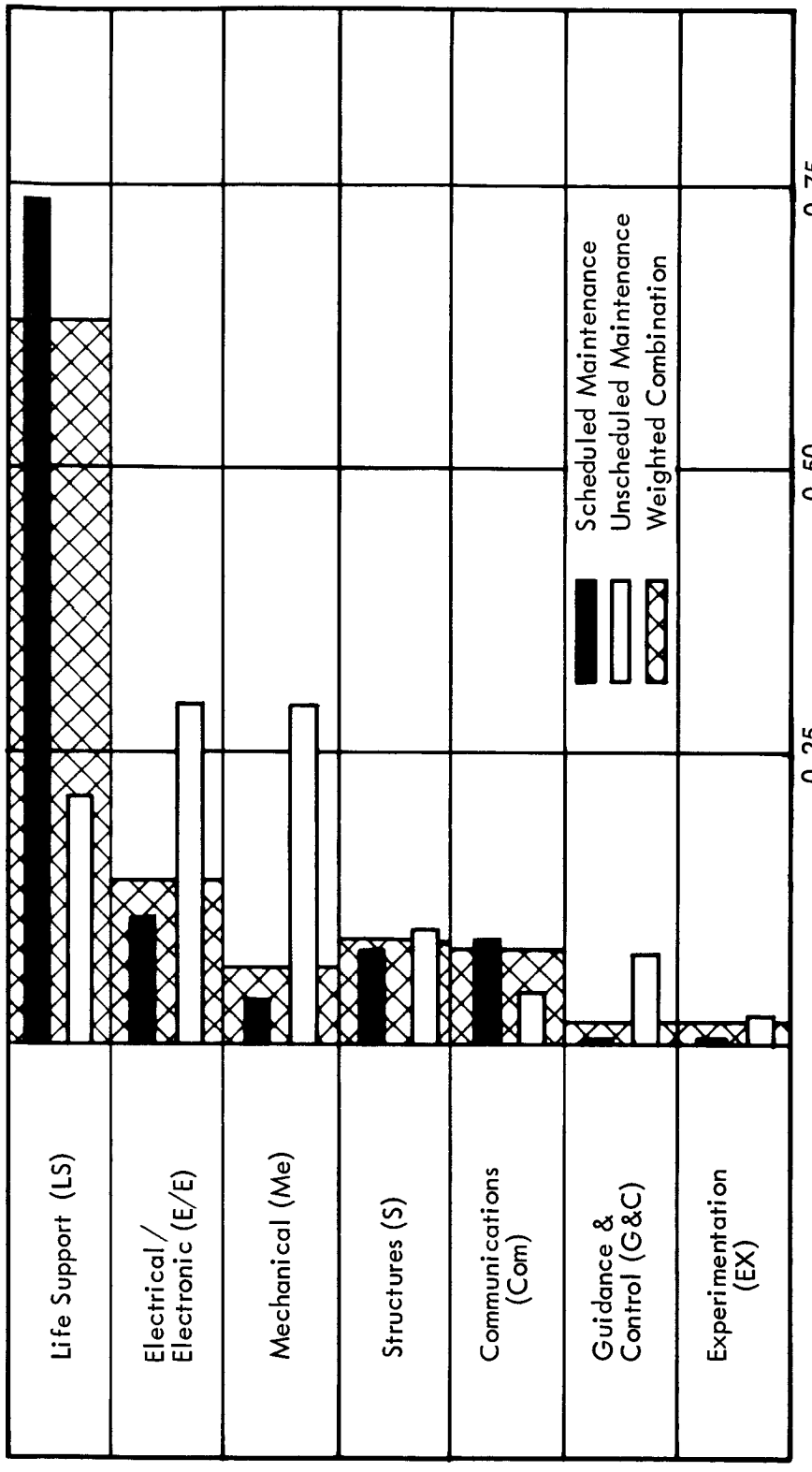


Figure 3.3-7: DISTRIBUTION OF MAINTENANCE SKILLS REQUIREMENTS

actual experiment tasks, evaluation of experiment results and assuring adequate completion of experiment, and recovery, checkout, refurbishment, and restowage of equipment. Gross estimates were made of the man-minutes per day to accomplish these tasks for each of the postulated experiments during each phase of the mission. Table 3.3-5 summarizes the task requirement estimates for two missions, a 1982 Mars opposition landing mission and a 1980 Venus short-stay capture mission. The in-transit phases of the mission are divided into the different coast periods between major propulsive and correction maneuvers. The Mars orbit stay is divided into four periods: the first corresponding to the time between orbit insertion and MEM descent, the second and third covering the period while the MEM is on the surface, and the fourth period representing the time between MEM ascent and rendezvous and the space vehicle's departure from the planet orbit. The Venus mission is similarly divided. For Mars surface experimentation, a requirement of 8 hours per day per man was assumed.

Specific Mars surface operations depend on the characteristics of the site selected as well as on the biological and geological "finds" during exploration. The following activities are the predominant ones if the primary emphasis is on life detection: (1) features of microscopic life including a study of the habits, environment, excrement collection, food and water sources, and photography; (2) selection of soil samples for microscopic life studies, estimation of water availability, visibility, environment, and microscopic slide preparation; (3) collection of sediments for fossil studies and microscopic studies of selected surface materials for cell structure analysis; (4) age-dating using radioactive carbon 14; and (5) emplacement of weather station.

The following activities dominate should initial investigation reveal the total lack of evidence for biological activity: (1) photography of surface features; (2) emplacement of weather station; (3) sediment formation studies through photography and analysis for formative mechanisms and age dating; (4) collection of samples representative of the sites, minerology, geology, and chemistry; (5) water or moisture availability; and (6) emplacement of seismic array and the application of coring techniques.

3.3.5.2 Skill Requirements

The skills required to perform the experiment tasks were identified in the analysis of Part 2 of this volume. Those skills were grouped into five categories: Physics, Astrophysics, Geology, Communications and Electronics, and Bioscience. The grouping of experiment skill requirements is presented in Table 3.3-6 and the distribution is given in Figure 3.3-8.

3.3.6 CREW CARE

The majority of crew care tasks will be those routine personnel operations involving each crew member, each day, to keep himself physically and mentally fit for performing his mission assignments. Special care may be required periodically in the form of professional medical or

Table 3.3-5: POSTULATED EXPERIMENT TASKS REQUIREMENTS

Mission Phase		Average Experiment Requirements		Total Experiment Requirements			
				Mars 1982 Opp.		Venus 1980 Short	
		Experiment Time (hr/day)	Manhours Required (manhr/day)	Elapsed Time (days)	Total Manhours	Elapsed Time (days)	Total Manhours
In-Transit	OUTBOUND TRIP						
	First Coast	3.3	3.3	5	16.5	5	16.5
	Second Coast	4.8	6.8	20	136.0	20	136.0
	Third Coast	5.0	8.7	245	2131.5	85	688.5
	Fourth Coast	4.0	6.5	20	130.0	20	130.0
	INBOUND TRIP						
	First Coast	4.0	6.5	5	32.5	5	32.5
	Second Coast	4.8	6.8	20	136.0	20	136.0
	Third Coast	5.0	8.7	165	1435.5	245	2131.5
	Fourth Coast	3.3	3.3	20	66.0	20	66.0
	SUBTOTAL	-	-	500	4084.0	420	3337.0
Mars Orbit	1st--5th Day	6.3	10.3	5	51.5		
	6th--20th Day	6.6	11.8	15	177.0		
	21st--35th Day	6.6	11.8	15	177.0		
	36th--40th Day	5.5	9.5	5	47.5		
	SUBTOTAL	-	-	40	453.0		
Mars Surface	1st--30th Day	8.0	*24.0	**30	720.0		
	SUBTOTAL	-	-	30	720.0		
Venus Orbit	1st--5th Day	19.9	37.3			5	186.5
	6th--20th Day	15.0	24.3			15	364.5
	21st--35th Day	17.5	24.8			15	417.0
	36th--40th Day	11.8	25.0			5	125.0
	SUBTOTAL	-	-			40	1093.0
	TOTAL	-	-	540	5077.0	460	4430.0

*Based on postulated three-man MEM crew.
**Occurs simultaneous with orbit experimentation between 6th and 35th days.

Table 3.3-6: EXPERIMENT CREW SKILL REQUIREMENTS

Skill Category	Skills	
Physics	Optics IR & UV visible Spectral analysis	Photographic analysis Mars spectrometry Nuclear instrumentation
Astrophysics	Astronomy Planetology Solar physics	Interplanetary physics Nuclear physics Atmospheric physics and meteorology
Geology	Geology Seismology	Physical chemistry X-Ray defraction
Communications & Electronics	Radar image interpretation Radio wave propagation Electronic equipment	
Bioscience	Biology Physiology	Botany Radio biology

psychiatric treatment. However, it is assumed that all crew members will be selected with appropriate physical characteristics and compatible social traits such that special treatment will be minimized.

3.3.6.1 Task Requirements

Routine personnel operations include such things as sleep, nutrition, personal care, leisure, physical conditioning, and training. Added breakdown of these items is presented in Table 3.3-7. Estimates of the amount of time that should be allocated for these items in crew planning and sizing analyses have varied considerably in previous studies, varying from 12 to about 15.5 hours per man day. Table 3.3-8 summarizes some of those estimates. The estimates used in this study total 13.3 hours per man per day and are shown in the last column of Table 3.3-8. An exception is made during the planet stay time for all short-stay missions (i.e., 40 days) by deleting the times allowed for training, leisure, and physical conditioning and reducing slightly the times allocated to eating and personal care. Ten hours was allowed during these planet stay times.

Special-care tasks would only be required when a crewman needs medical or psychiatric treatment or is sick, injured, or otherwise incapacitated. It is impossible, at this time, to accurately predict the tasks that might actually be required in this area. Through long-term orbital testing, the sensitivities of man to the space environment and to long-term confinement will be identified and a better idea of what will be required will be derived. It is safe to assume that some professional medical treatment capability is essential.

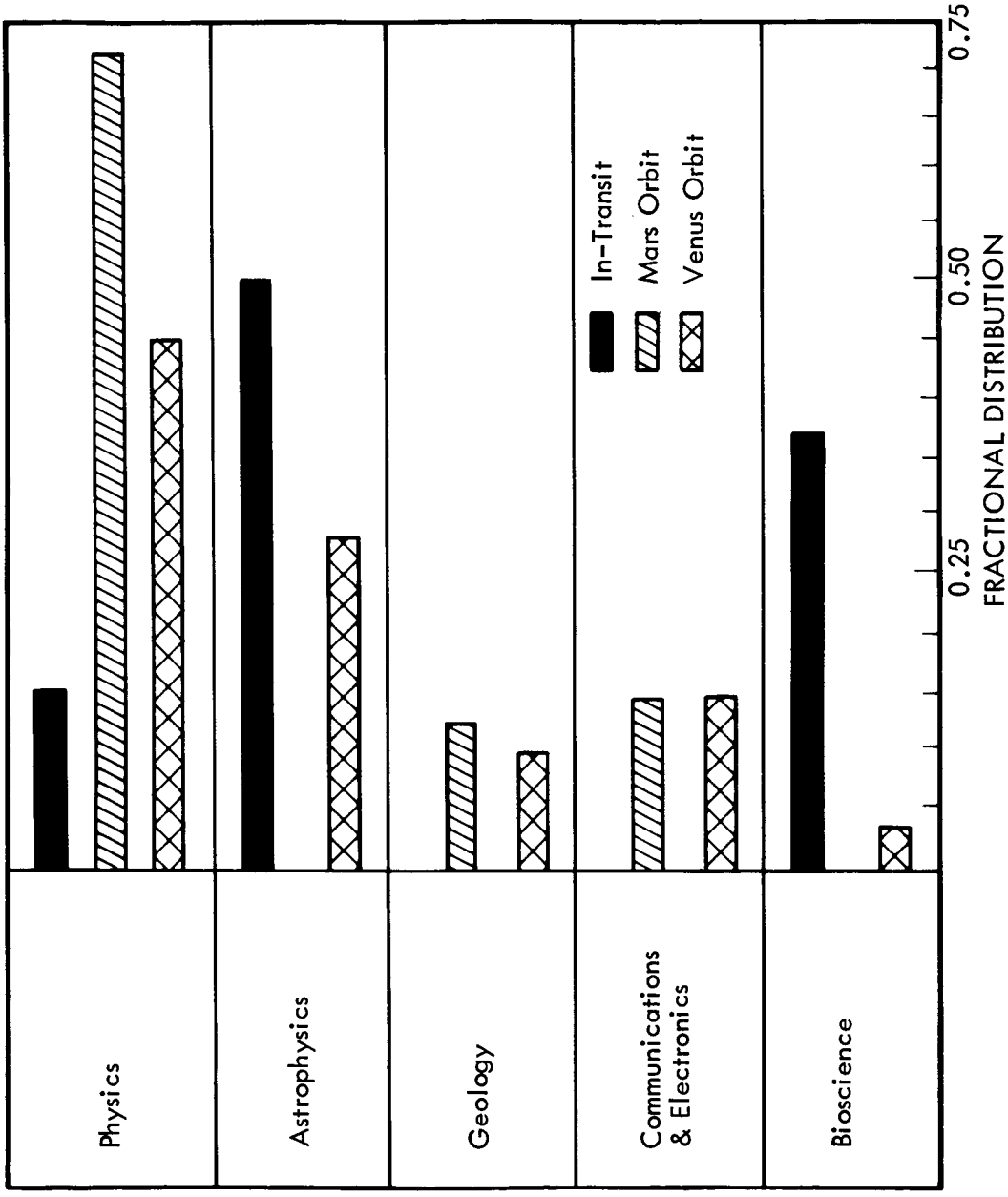


Figure 3.3-8: DISTRIBUTION OF EXPERIMENT SKILLS

Table 3.3-7: ROUTINE PERSONNEL TASKS

Sleep	Preparation Sleep Wake and Dress (24-hour individual cycles should include one or two uninterrupted sleep periods.)
Nutrition	Food Preparation Eating Clean Up
Personal Care	Personal Cleansing Dental and Oral Hygiene Shaves, Hair Cuts,and Nail Cutting Waste Elimination
Leisure	Relaxation Recreation
Physical Conditioning	Condition Assessment Exercise Gravity Conditioning
Training	Skill Reinforcement Expansion of Skills Emergency Drills

3.3.6.2 Skill Requirements

It is assumed that each of the crewmen selected for the interplanetary missions will have had some space operations experience in Earth orbit and will be adequately trained in the general skills necessary for the routine personnel tasks. For medical treatment, medical skills comparable to a general physician's will probably be required. With the crew being confined to the space vehicle for such a long period of time, psychiatric and psychological skills at a professional level may be required to detect problems as they develop and provide the necessary therapy.

Table 3.3-8: SUMMARY OF ESTIMATIONS OF ROUTINE PERSONNEL OPERATIONS (HRS/DAY/MAN)

	LTV-AOLO Study Rpt. No. 00.368 4 May 1964	NAA-Manned Earth Orbiting Station Study, SID63-1392 18 November 1963	Douglas-Manned Exploration in Unfavorable Time Period, SM-45581 February 1964	GD/A-A Study of Manned Inter-planetary Missions GD/A-AOK64-006-5 1 July 1964	Boeing-OLF Study D2-82559-2 October 1965	Boeing Maintain-ability of Long Duration Manned Space Flights, NAS2-3705, 1966	Douglas-Report on a System Compari-son & Selection Study of a MORL, SM-44599, Sept. 63	Recommended Estimates for Study
Sleep	8.0	8.0	7.0	7.0	8.0	7.0	8.0	7.0
Nutrition	2.5	1.0	2.7	?(4)	2.5	3.0	1.8	2.5
Personal Care	1.5	1.0	3.5(2)	?(4)	1.5	3.5(6)	0.7	1.0
Leisure	1.0	2.0(1)		4.0	2.2(5)		1.5	1.0
Physical Conditioning	1.0		2.0(3)	1.0			3.5	1.5
Training	-	-		-	0.3	-	-	0.3
Total	14.0	12.0	15.2	?	14.5	13.5(7)	15.5	13.3*

1) This is divided into 1 hr physical recreation and 1 hr mental recreation.
2) This includes bathing, grooming, recreation, toilet, hobbies, etc.
3) This includes proficiency maintenance, skill acquisition, and physical conditioning.
4) Not identifiable in the study.
5) This is divided into 0.2 hr crew assessment and 2 hr relaxation and conditioning.
6) This is divided into 0.5 hr defecation, 0.5 hr urination, 0.5 hr exercise, and 2 hr personal time.
7) Allowable deviations over a 72-hour cycle would vary between the extremes of 10 hr/man/day and 16 hr/man/day.

*For a 40-day planet stay time, this is reduced to 10 hours per day.

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4.0 MISSION SUCCESS AND CREW SAFETY ANALYSIS

Two analyses were conducted to provide an assessment of mission success probability and crew safety and to provide estimated reliability requirements. First, a reliability analysis was made which defined system reliabilities at the module level (mission module, propulsion modules, etc.) and applied these to the analysis of mission success probability for a representative Mars mission. For comparison, two alternative convoy modes were also analyzed. Secondly, a trajectory analysis was made for six representative missions to determine the outbound leg abort capability for the recommended system with no extra provisions for abort. Using the reliability data, estimates were then made of the probability of successful abort from given points in the mission profile.

4.1 MISSION SUCCESS PROBABILITY

For this analysis mission success is defined as completion of each phase of the planned mission and return of the astronauts to Earth. The system is composed of the following nine major elements:

1) Mars Excursion Module (MEM)
2) Mission Module (MM)
3) Earth Entry Module (EEM)
4) Propulsion Module 1 (PM-1)
5) Propulsion Module 2 (PM-2)
6) Propulsion Module 3 (PM-3)
7) Propulsion Module-Outbound Midcourse Correction
8) Propulsion Module-Mars Orbit Trim
9) Propulsion Module-Inbound Midcourse Correction

Previous studies of manned interplanetary missions show that the module reliability values are a function of the weight allocated for redundancy and spares. In this study, a parametric approach was used in which a detailed analysis was made for the mission module, and reliabilities for the other modules were derived by a process of ranking them relative to the mission module.

4.1.1 MISSION MODULE RELIABILITY

Boeing's MARCEP (Maintainability and Reliability Cost Effectiveness Program) was used to establish reliability estimates for the mission module. This program determines the best combination of parallel or standby redundancy and spares required to achieve a given system reliability in a cost-effective manner. It examines each component of a basic system to determine the optimum (least weight) method of adding another system

component of the same type, taking into consideration allowable down-time, repair time, failure rate, and criticality of the component in the overall system. In an iterative process, components are selected and added to the system in the optimum manner until the required system reliability is achieved or until a defined cost constraint (weight, volume, or dollars) has been reached. In the MARCEP run for this study, no cost constraint was applied and the computation was run to provide curves of system reliability versus added weight. Details of the analysis are reported in Boeing Document D2-113204*. Figure 4.1-1 shows the resulting plot of mission module reliability for a 1986 Mars opposition mission of 480-day duration. Two points were selected from this curve for use in calculations of the reliabilities of other modules and in subsequent assessments of mission success probabilities. The higher value, 0.985 was used as the design point.

4.1.2 RELIABILITY ALLOCATION

Relative unreliabilities of the system elements were qualitatively ranked based on the following factors which were identified as influencing reliability:

1) Complexity
2) State of the art
3) Operating time (storage and operating)
4) Environment
5) Reliability growth.

A chart such as shown in Table 4.1-1 was distributed to members of the study team with instructions to rank the system elements according to the effects that these factors had on reliability. These factors were then combined at the module level to yield an estimate of the fraction of total system unreliability attributable to each system module as shown in Table 4.1-1.

Using the two values previously noted for the mission module reliability, the corresponding values for the other major elements were calculated. The resultant module reliability estimates are shown in Table 4.1-2.

4.1.3 SINGLE-VEHICLE MODE MISSION SUCCESS PROBABILITY

For the single-vehicle mode used throughout the study, probability of mission success is plotted as a function of mission time for two levels of reliability in Figures 4.1-2 and 4.1-3. With respect to the analysis model, no catastrophic-type failures are considered; that is, no module failure shall directly affect the success or failure of another module, and the model is a single chain of events.

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*Boeing Document D2-113204, *Maintainability of Manned Spacecraft for Long-Duration Flights*, NASA Contract NAS2-3750, July 1967.

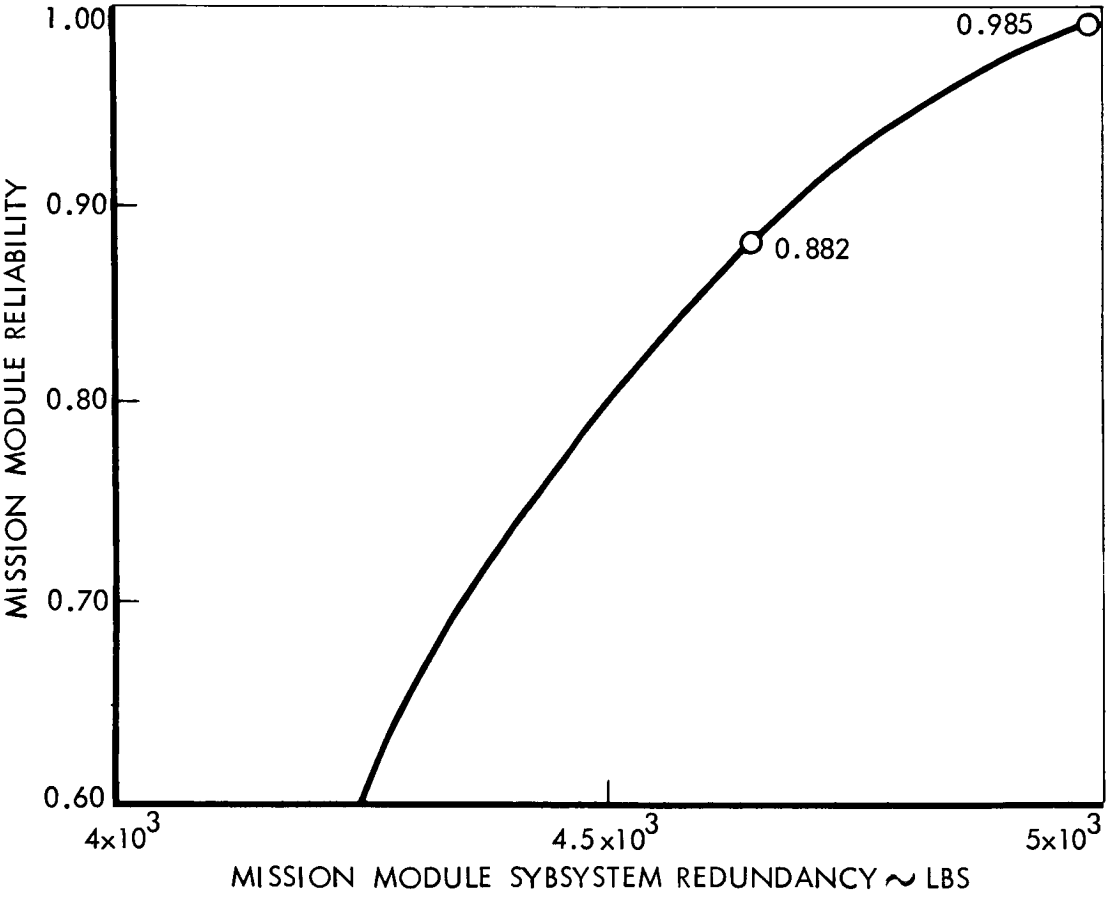


Figure 4.1-1: MISSION MODULE RELIABILITY — 1986 MARS OPPOSITION MISSION (480-day Trip Time)

Table 4.1-1: FACTORS INFLUENCING RELIABILITY

System Elements	Complexity	State of the Art	Operating Time	Environment	Reliability Growth	Fraction of Total
MEM	8	7	2	5	6	0.290
MM	10	8	10	1	7	0.483
EEM	5	6	1	8	5	0.104
PM-1	1	9	1	3	9	0.021
PM-2	1	9	2	3	9	0.042
PM-3	1	9	2	3	9	0.042
PM-Outbound Midcourse Correction	1	2	4	3	3	0.006
PM-Orbit Trim	1	2	4	3	3	0.006
PM-Inbound Midcourse Correction	1	2	4	3	3	0.006
Total						1.000

Since various alternative modes are often postulated for the purpose of improving mission success probability or the probability of crew survival, the analysis was extended to two alternative convoy modes, discussed in the following paragraphs.

4.1.4 CONVOY MODE--OVAM

General Dynamics/Astronautics has studied several convoy modes*. After a review of their work, their "orbital vehicle assembly mode" (OVAM) was selected for comparison with the single-vehicle mode. This one was judged to be the best of the several modes considered in the GD/A study. In this convoy mode both vehicles are completely assembled and checked out in Earth orbit before Earth departure. This means that all remating and assembly operations during the mission are eliminated unless a failure occurs. The two vehicles that make up the OVAM consist of the following:

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*GD/A-AOK64-006-6, *A Study of Manned Interplanetary Missions*, Volume 6--Operations Studies, NASA Contract NAS8-5026, General Dynamics/Astronautics, July 1964.

Table 4.1-2: RELIABILITY ALLOCATION

System Element	Reliability Estimate	
	High	Low
Mars Excursion Module	0.9910	0.9290
Mission Module	0.9850*	0.8816*
Earth Entry Module	0.9968	0.9745
Propulsion Module 1	0.9993	0.9949
Propulsion Module 2	0.9987	0.9897
Propulsion Module 3	0.9987	0.9897
Propulsion Module - Outbound Midcourse Correction	0.9998	0.9985
Propulsion Module - Inbound Midcourse Correction	0.9998	0.9985
Total (Single Mode)	0.969	0.774

*The mission module reliability served as the basis for allocation of system reliability to constituent modules.

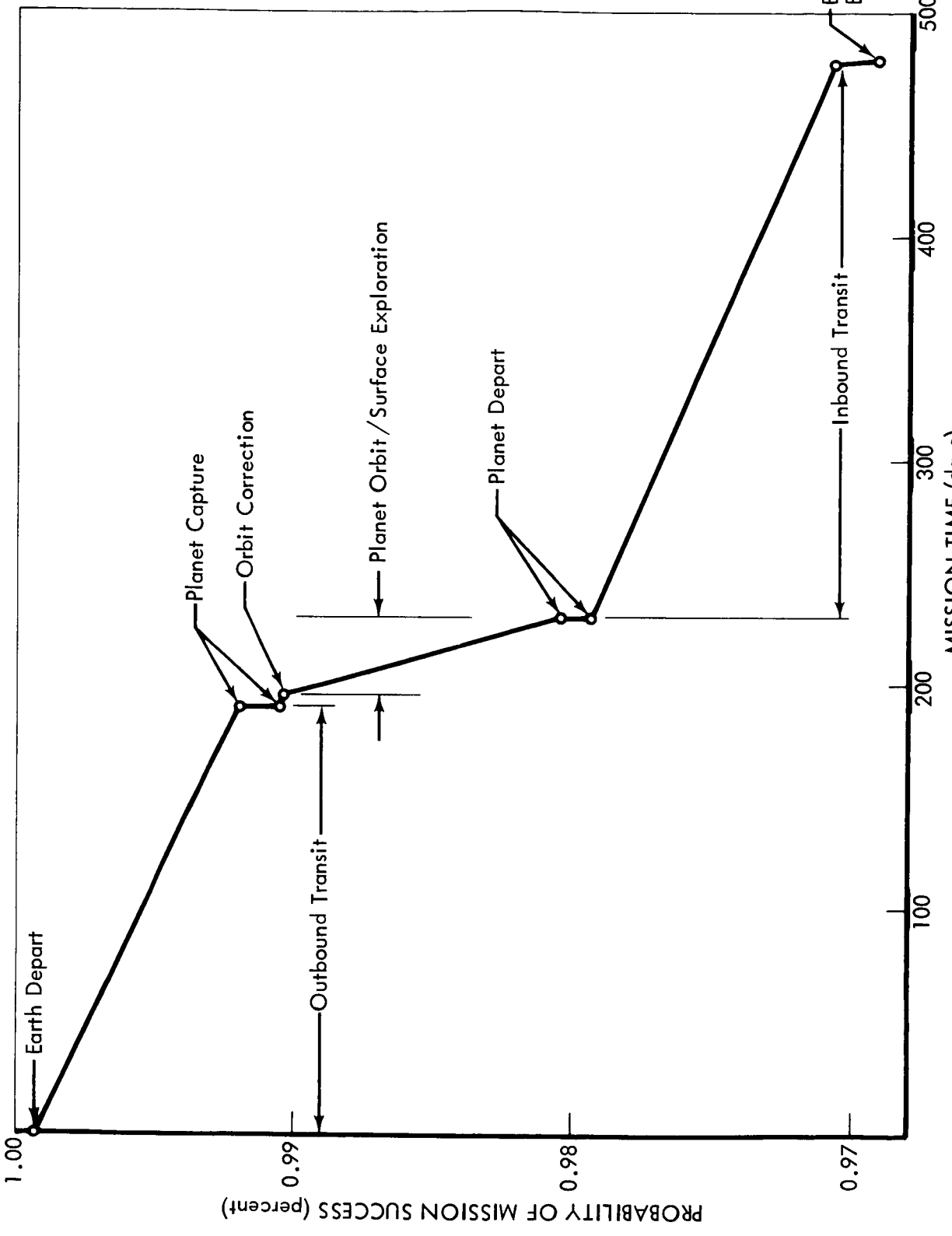


Figure 4.1-2: PROBABILITY OF MISSION SUCCESS — SINGLE-VEHICLE MODE (0.9850 MM Reliability)

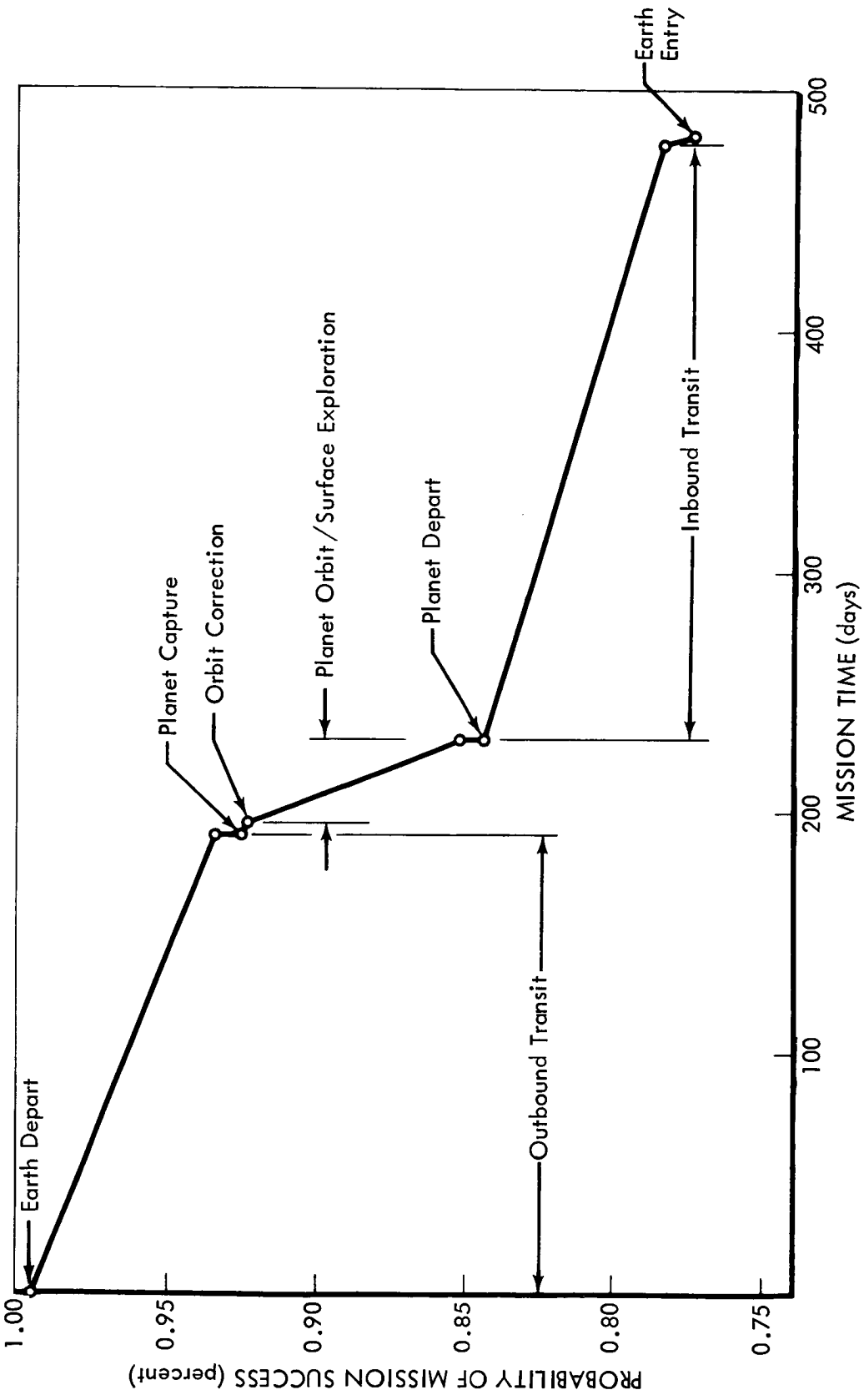


Figure 4.1-3: PROBABILITY OF MISSION SUCCESS — SINGLE-VEHICLE MODE (0.8816 MM Reliability)

1) Vehicle A (Manned Vehicle) - The payload of Vehicle A consists of an Earth entry module (EEM), the mission module (MM), one-third of the scientific package, and the probes.

2) Vehicle B (Unmanned Vehicle) - The payload of Vehicle B consists of an EEM, the Mars excursion module (MEM), and two-thirds of the scientific package.

This method of splitting the payload between the two vehicles closely parallels the payloads suggested by GD/A. This distribution was chosen because it gives nearly equal payload weights for the two vehicles.

The planet capture and departure propulsion modules (PM-2 and PM-3) for both vehicles were sized such that they would be able to impart the necessary ΔV's to the payload of Vehicle A since it is the heavier of the two payloads. The Earth departure propulsion module, PM-1 (A), was sized to give ΔV_1 to Vehicle A's payload. PM-1 (B) was sized to be able to impart ΔV_1 to the payload of Vehicle B. This method of sizing was chosen to keep the convoy mode (OVAM) IMIEO to a minimum for comparison purposes, since the capability of interchanging the PM-1's of the two vehicles is not essential. The modules used in both vehicles are of the same design as for the recommended single vehicle.

The mission operation sequence flow is shown in Figure 4.1-4. The time marks shown in Figure 4.1-4 are for the 1986 Mars opposition mission used to evaluate the convoy mode. The main differences in the operation for a single-vehicle mode and a convoy mode are as follows:

1) It is necessary for the two vehicles to rendezvous immediately after disposing of their PM-1's.

2) During the coasting phases of the mission, any needed repair to the equipment or modifications of the experiments on Vehicle B is done by sending men from Vehicle A to Vehicle B. Otherwise, there would be no reason for men to go aboard Vehicle B. This is desirable in light of the experiments being conducted from Vehicle B and the possibility of a man on board disturbing them.

3) The two vehicles must rendezvous after their orbit trim maneuver in Mars orbit.

4) The men going to the surface in the MEM must transfer from Vehicle A to Vehicle B.

5) The MEM rendezvous with Vehicle B when it leaves the Martian surface, and the Mars samples are transferred to the EEM of Vehicle B. This is done to help prevent contaminating the samples. Then the men transfer to Vehicle A taking approximately 100 pounds of samples with them.

6) The two vehicles must rendezvous after disposing of their PM-3's.

7) During the last coasting phase of the return trip, approximately 200 pounds of samples are transferred from Vehicle B to Vehicle A so that in case of the loss of the unmanned EEM not all the samples are lost also.

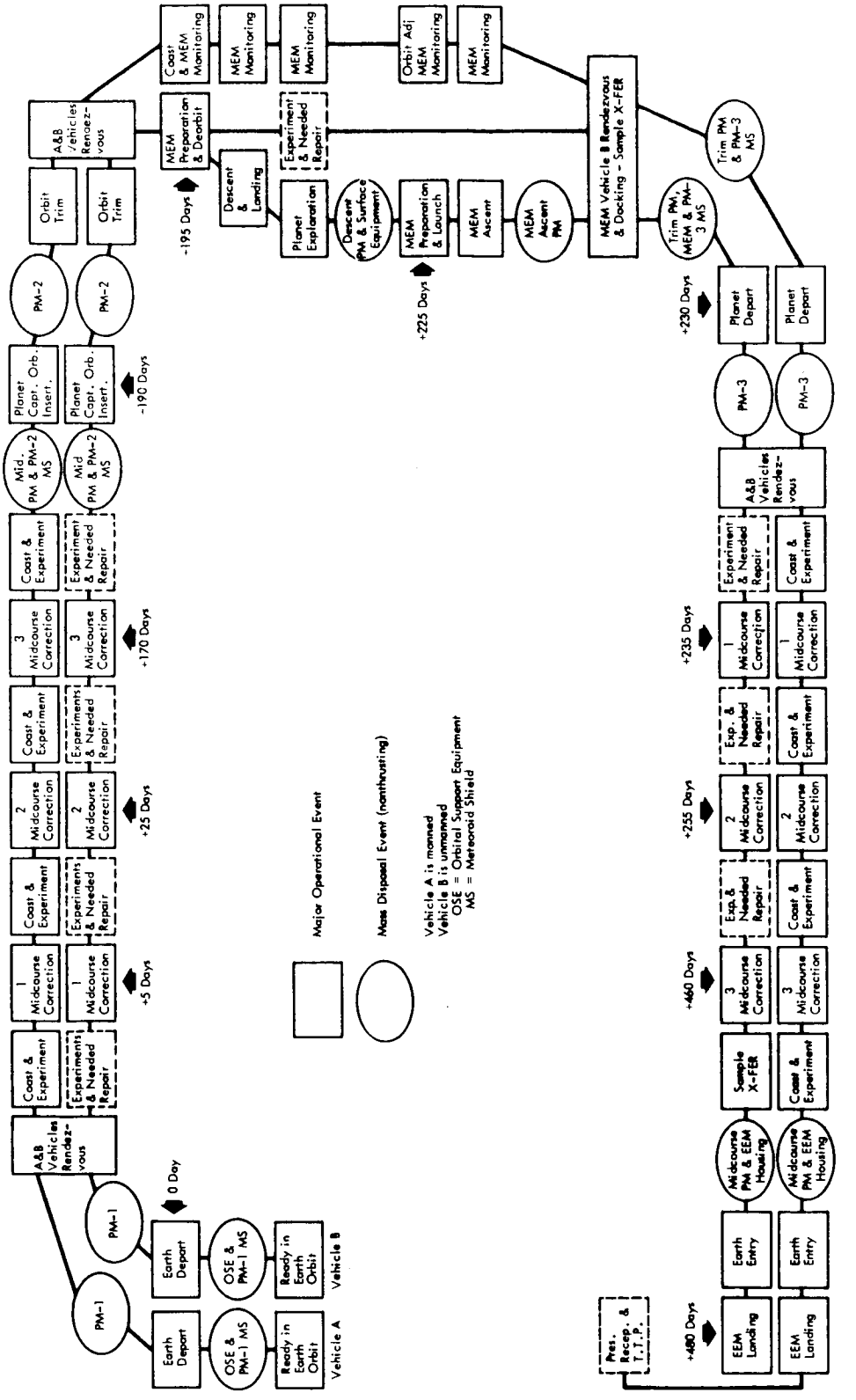


Figure 4.1-4: MISSION OPERATION SEQUENCE FLOW (CONVOY MODE)

8) The EEM of Vehicle B is controlled from the ground during Earth entry and landing.

The action to be taken in case of a module failure is shown in Table 4.1-3. In Action No. 1, the "MM Vehicle" is the manned vehicle whether this is Vehicle A, or Vehicle B after the MM is transferred to it. Action No. 5 is possible only if ΔV_3 is larger than ΔV_2 as is true for the 1986 Mars opposition mission used here. Action No. 6 is applied only to the orbit trim maneuver which, when it is eliminated, necessitates a change in chosen landing site.

4.1.5 CONVOY MODE (S + C)

A second convoy mode was analyzed when it was found that the above described OVAM mode offered essentially no improvement in mission success probability. This convoy mode consists of sending Vehicle S which is the single-mode vehicle and is identical to the recommended vehicle described in Volume IV in company with a Vehicle C. The payload of Vehicle C consists of an EEM, a MEM, two-thirds of the scientific package, and probes i.e., two Mars excursion modules are carried to Mars.

All of the PM's on Vehicle C, except PM-1, are sized the same as the PM's on Vehicle S. PM-1(C) was sized to impart ΔV_1 to Vehicle C.

The mission operation sequence for this convoy mode is the same as the one shown in Figure 4.1-4 with the following exceptions:

1) The MEM preparation would take place on board Vehicle S.
2) The returning MEM would rendezvous and dock with Vehicle S and there would be no sample transfer.
3) There would be no sample transfer during the last coasting phase of the inbound leg.

If S's are substituted for A's, and C's substituted for B's, the actions taken in case of a failure in this convoy mode are the same as shown in Table 4.1-3 with the following exceptions:

1) Action No. 1 should read MM vehicle continues as planned.
2) During the capture orbit phase, if the MEM(S) should fail in orbit, MEM(C) would be used for surface exploration. If MEM(S) fails on the surface then MEM(C) is used for a rescue vehicle.
3) Action No. 7 would not have to be used in case of any C vehicle failures.

4.1.6 CONVOY MODE COMPARISONS

Probability of mission success for the two convoy modes are compared with the single-vehicle mode in Figures 4.1-5 and -6. These plots display the probabilities of successful equipment performance from Earth depart up to completion of the indicated mission phases. Figure 4.1-5 reflects high reliability values for the various system modules (compatible with 0.9850 reliability for the mission module). The lower probabilities of

Table 4.1-3: OPERATIONAL MODES FOR MODULE FAILURES

Flight Phase	Module Not Usable or Doesn't Function	Action Number
Earth Departure	PM-1(B)*	1
	PM-1(A)	3
Outbound Leg	Midcourse Correction PM(B)	1
	Midcourse Correction PM(A)	4
	PM-2(B)	5
	PM-2(A)	5 & 4
Planet Capture	PM-2(B)	5
	PM-2(A)	5 and after deployment of MEM & probes 4
Capture Orbit	Orbit Trim PM(B)	6
	Orbit Trim PM(A)	6 and after deployment of MEM & probes 4
	MEM	1 and deployment of probes
	PM-3(B)	7
	PM-3(A)	7 & 4
Planet Depart	PM-3(B)	2
	PM-3(A)	8
Inbound Leg	Midcourse Correction PM(B)	7 & 2
	Midcourse Correction PM(A)	7 & 4
	EEM(B)	7 & 2
	EEM(A)	7 & 4
Earth Entry	EEM(B)	2
	EEM(A)	8

Explanation of Action Numbers

Action No. 1, Mission module vehicle continues as planned with mission changed to an orbital one.

Action No. 2, Vehicle A continues as planned.

Action No. 3, Vehicle B continues into Mars orbit. Original mission is scrubbed.

Action No. 4, Payload B is taken off Vehicle B. The mission module is taken off Vehicle A and placed on Vehicle B.

Action No. 5, PM-2 is jettisoned and PM-3 is used to put vehicle into orbit around Mars.**

Action No. 6, Maneuver deleted.

Action No. 7, All samples transferred to Earth entry module of operable vehicle.

Action No. 8, Rescue operations initiated from Earth, if possible.

*(B) denotes module of Vehicle B
(A) denotes module of Vehicle A
** 1986 Mars Opposition Mission

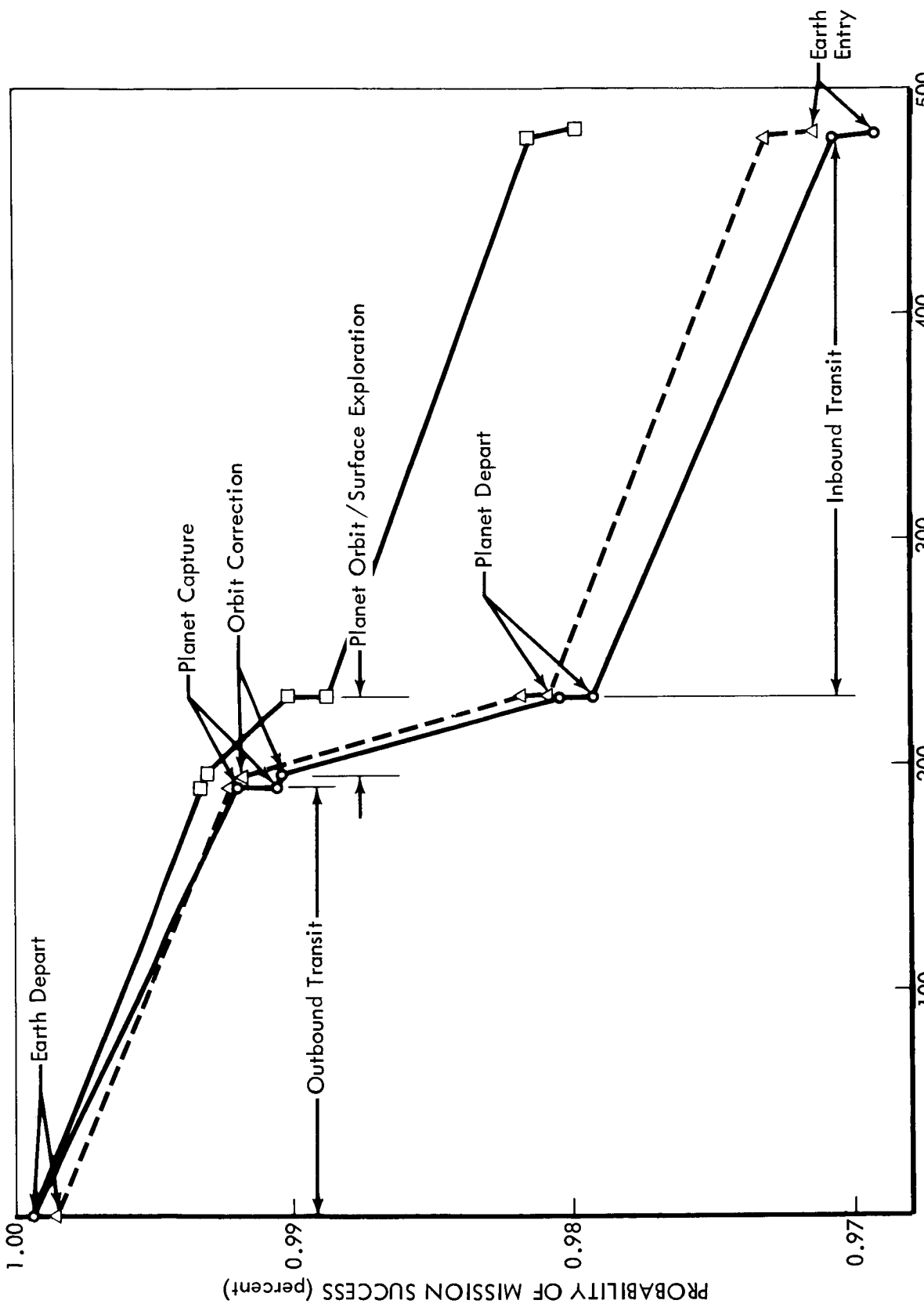


Figure 4.1-5: HAZARD PLOT (0.9850 MM Reliability)

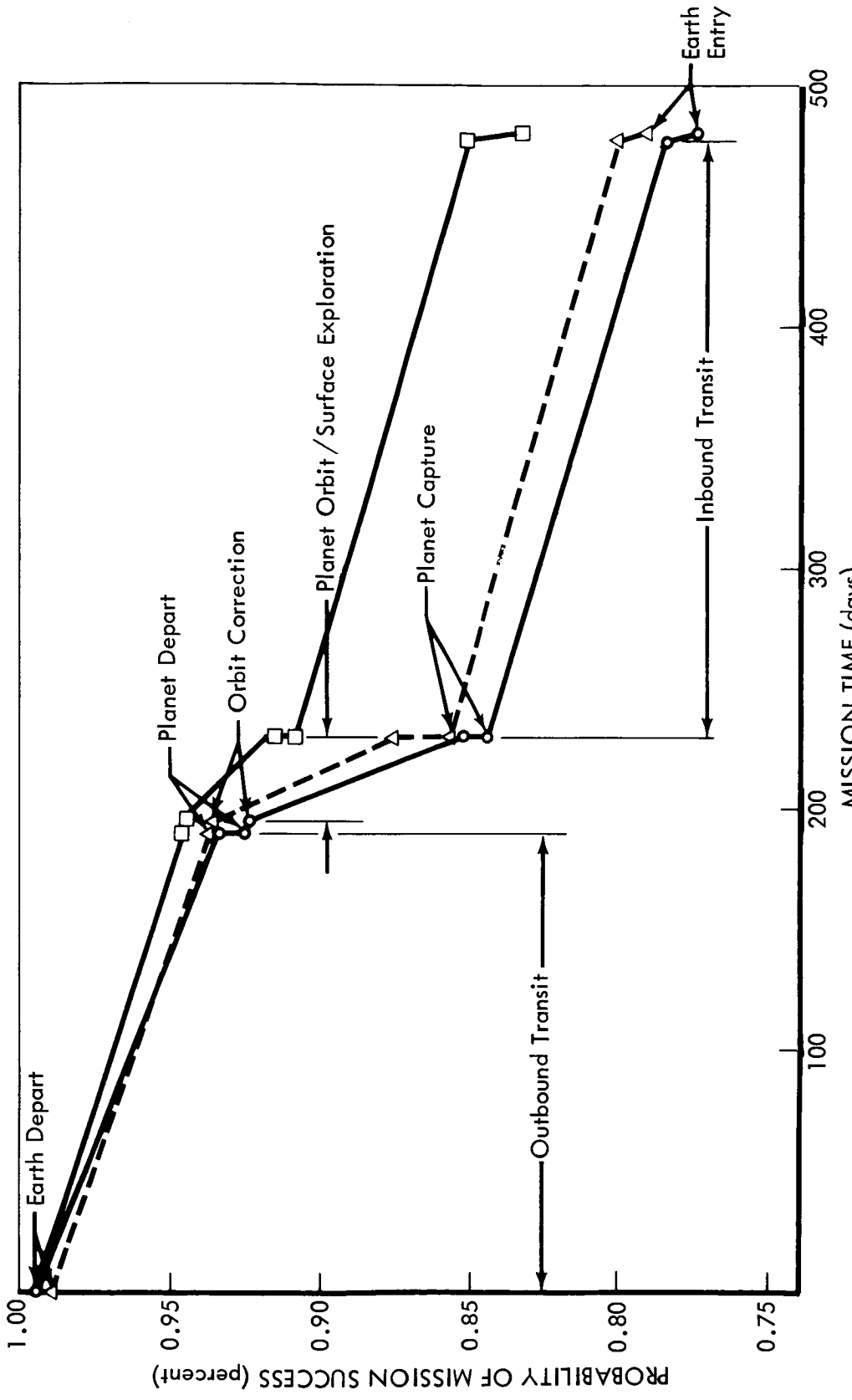


Figure 4.1-6: HAZARD PLOT
(0.8816 MM Reliability)

Figure 4.1-6 were calculated using lower module reliabilities compatible with a mission module reliability of 0.8816. The OVAM mode shows very little improvement over the single-vehicle mode, but the addition of a second MEM raises the probability of mission success substantially.

The basic motivation for the convoy mode in this case is improvement of the probability of crew survival, rather than of overall mission success. Although these measures are related, a more specific measure of crew survival is desirable for the comparison. A detailed safety analysis involves an integration over the complete mission profile, of the product of the probability of achieving a successful abort at a given time and the probability of reaching that point in the mission profile, and it was not practical to perform complete crew safety analyses in this study. However, some abort predictions were made to obtain a measure of safety. The probability of successfully returning the crew to Earth was calculated for the OVAM convoy mode and for the single mode, assuming that the mission was aborted at two different points in the mission profile: at successful Mars' orbit attainment and at successful Mars landing. Another way of stating the problem is: "given that the mission arrives successfully at this point in the planned mission, what is the probability of getting back safely if the mission is aborted at this point?" Although the answer is not a statement of the overall probability of crew survival, it is a valid basis for comparison. On this basis, the following results were obtained:

Abort Point	Mission Mode	Probability of Successful Abort
In Mars Orbit	Single	0.9100
	OVAM Convoy	0.9186
After Mars Landing	Single	0.8932
	OVAM Convoy	0.9016

These numbers are based on the lower of the two sets of reliability estimates given in Section 4.1.1. The differences in these abort conditions is negligible for the two modes. The comparable values for the second convoy mode were not generated, but because of the presence of a second MEM in that mode, the probability of return from the Mars surface would be improved. An estimate of the upper limit for that case was about 0.92.

4.1.6.1 Methods of Analysis--Convoy Modes

For the convoy modes of transportation, the mission success evaluation model required estimates of module reliability by mission phase and in some instances, by mode in addition to the reliability data of Section 4.1.2. The mission that formed the basis of the calculations is depicted in Figure 4.1-7. This mission consisted of eight major phases as shown by the column headings. The associated time intervals are shown in the column headings. Also shown in this figure are percentages of total unreliability by major system element for each phase of the mission. These percentage allocations were based primarily on respective time length of each phase, environmental influences, and operational duty cycles. Table 4.1-4 shows a fractional breakdown of major element unreliabilities for operating versus nonoperating modes.

Table 4.1-4: FAILURE PROBABILITY DISTRIBUTION BY MODE

System Module	Fraction of Module Unreliability	
	Nonoperating	Operating
MEM	0.05	0.95*
MM	0.00	1.00
EEM	0.50	0.50
PM-1	---	1.00
PM-2	0.10	0.90
PM-3	0.12	0.88
PM-Outbound Midcourse Correction	0.005	0.995
PM-Orbit Trim	0.07	0.93
PM-Inbound Midcourse Correction	0.01	0.99

*0.95 fraction of operational unreliability (failure probability) is broken down as follows:

0.10 - descent to Mars surface
0.75 - during surface exploration
0.10 - ascent to orbit

It is assumed that failure during descent or ascent will be catastrophic but that rescue can be made following any failure occurring during surface exploration phase.

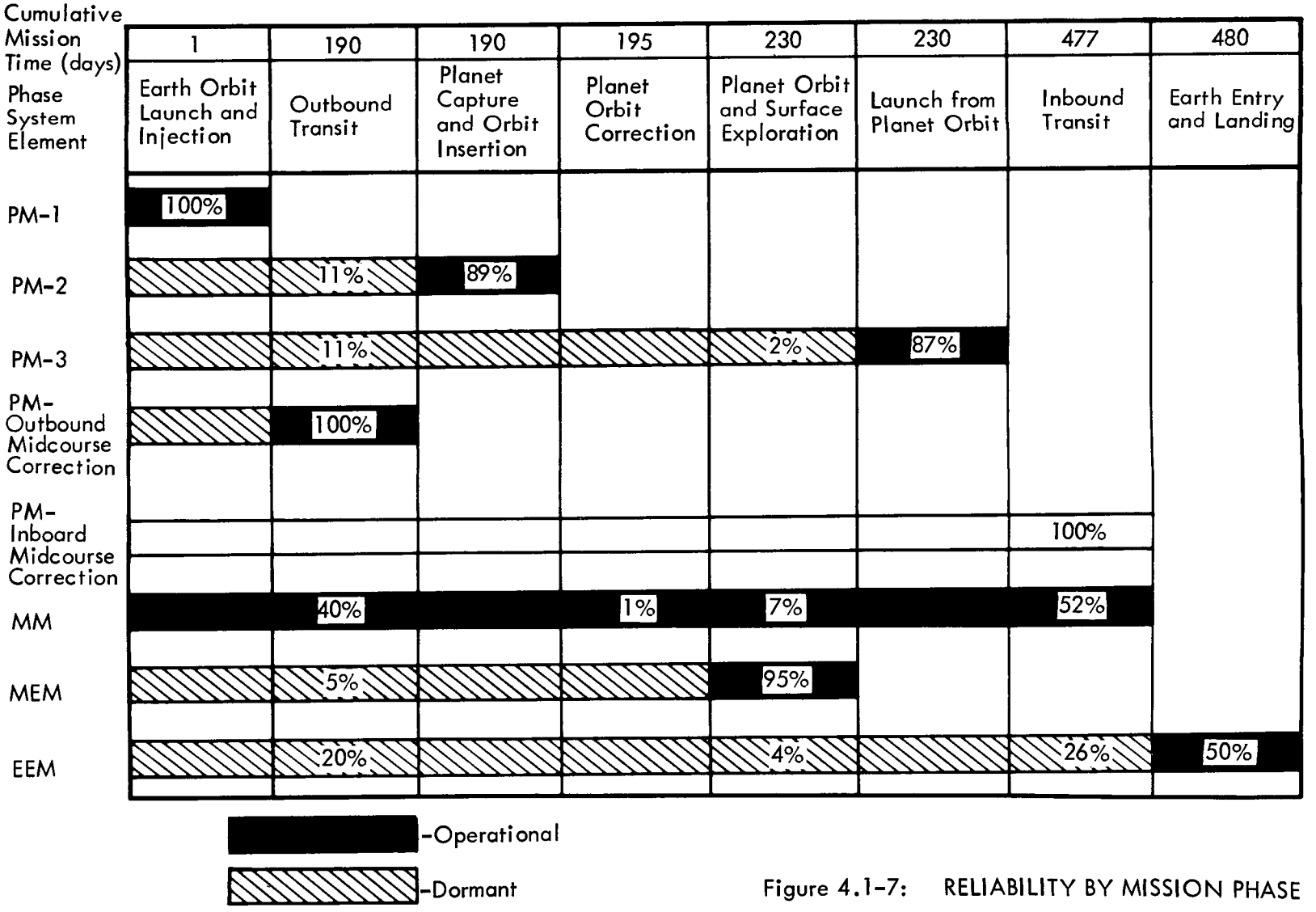


Figure 4.1-7: RELIABILITY BY MISSION PHASE

An additional allocation of MEM reliability was also required for use in the convoy mode analytical model. As annotated in Table 4.1-4, 75% of the failure probability for the MEM was assumed to result during the Mars surface exploration phase of the mission. The remaining portion of the operational failure probability was assumed to be equally distributed between the Mars descent and ascent phases.

Calculations for the probability comparison were made as shown in Figure 4.1-8 for the OVAM and Figure 4.1-9 for the (S+C). The models shown in these figures relate the reliability of the constituent system modules to the probability of mission success. These models take the form of success path diagrams which for each mission mode show the possible ways in which the mission can be concluded successfully. These success path diagrams show the complete logic involved in describing mission success; thus, they represent a Boolean equation in graphic form in each case. Since this is the case, there is no need to translate the diagrams into standard Boolean expressions, and the success probability calculations are made on the diagrams. Module reliabilities are noted in the appropriate blocks and were combined according to the logic represented by the diagram. Note that in reading these diagrams "and gates" are represented by a simple connection of blocks, while "or gates" are represented by branches in the connecting lines, and success is achieved when an arrow is reached. Charts of the same form were used to compute additional points along the mission profiles.

The following assumptions and ground rules were used in the reliability prediction model:

1) No catastrophic-type failures are considered. That is, no module failure shall directly affect the success or failure of another module.

2) There is no backup for failure of a PM-1 or PM-3 in use. However, in the OVAM and (S+C), it is assumed that the condition of these modules is monitored and if they fail prior to intended use, a backup may be available (contingent on probability that it has not failed and that the alternate vehicle has successfully maintained the prescribed flight path).

3) In the (S+C), the spare MEM is assumed to provide backup for failure of the primary MEM prior to scheduled use (subject to the same contingencies on Vehicle C performance as enumerated in 2). Also the additional MEM may serve as a rescue vehicle in the event that the primary MEM should fail in use in such a manner that rescue is possible.

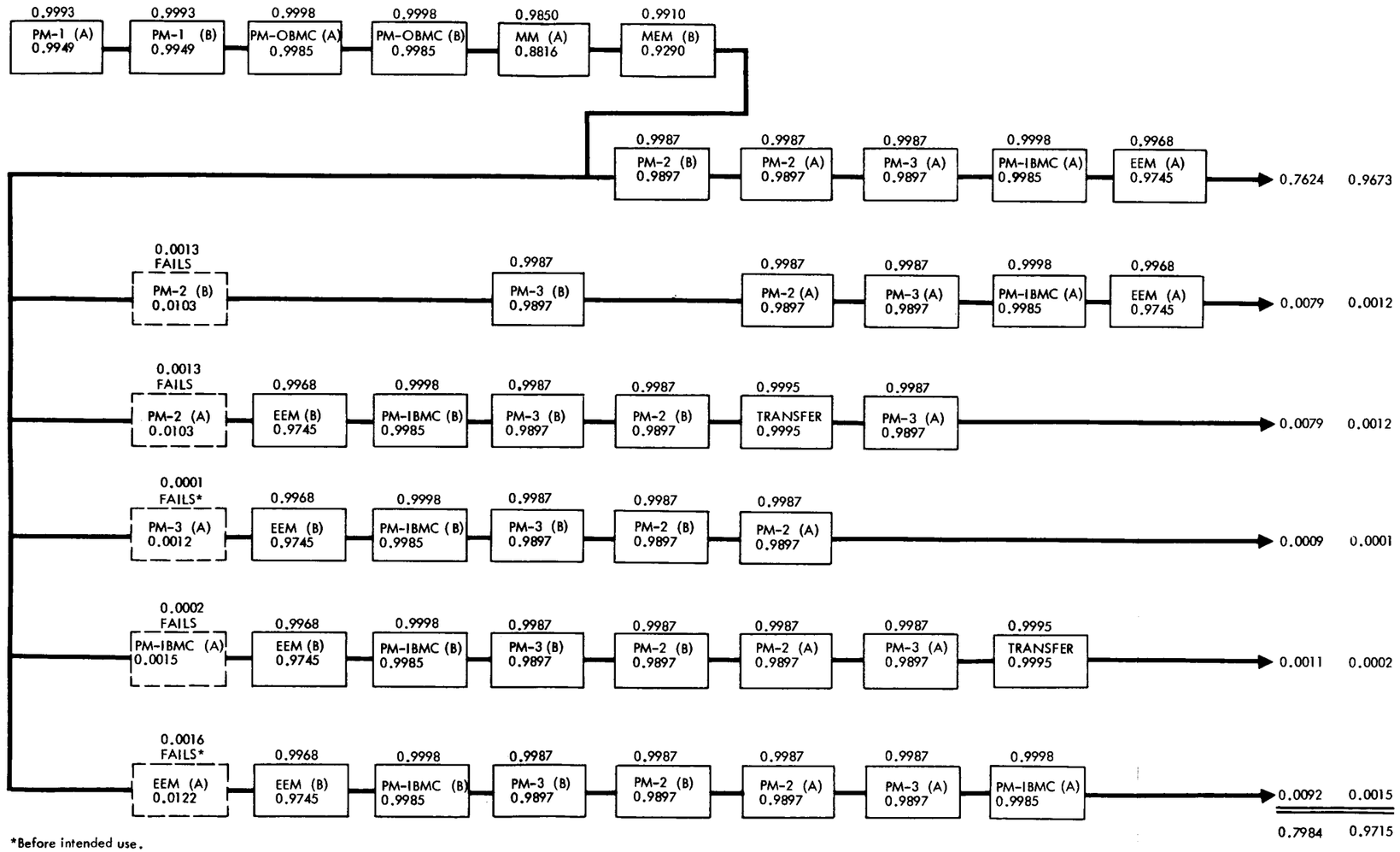
4) The probability that the MEM will fail in use such that rescue is not possible is assumed to be a function of the following only:

   a) MEM descent

   b) MEM ascent.

5) All MEM failures occurring on Mars' surface are assumed to be such that rescue can be made given availability of a functioning spare MEM.

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*Before intended use.

Fig. 4.1-8:A

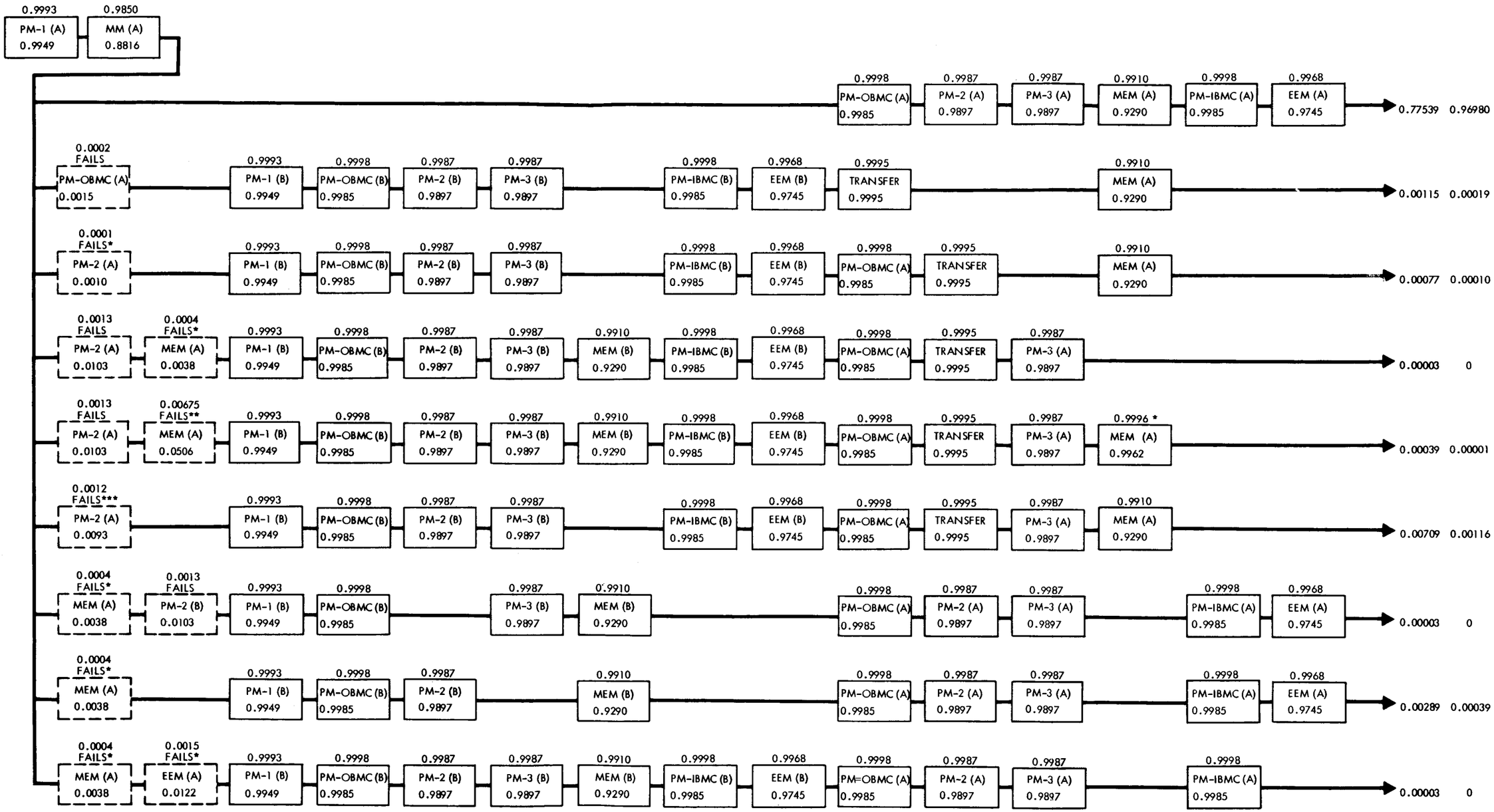
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Figure 4.1-8: B RELIABILITY MODEL OVAM

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* Before scheduled use.
** In use - such that rescue can be made with MEM (B).
*** In use.

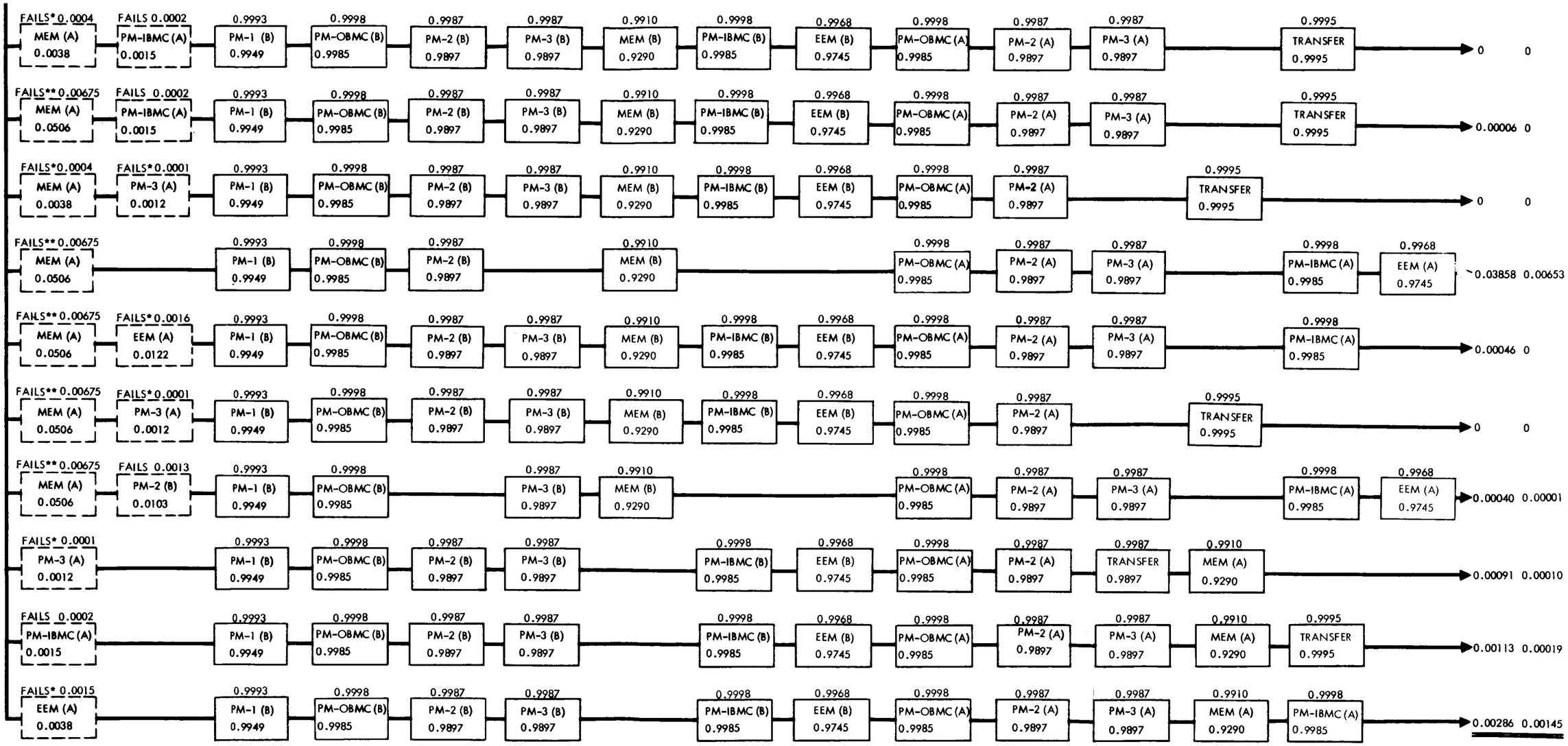
OBMC = Outbound Midcourse Correction
IBMC = Inbound Midcourse Correction

Fig 4.1-9: A
215
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Figure 4.1-9: RELIABILITY MODEL — S + C

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* Before scheduled use.
** In use - such that rescue can be made with MEM (B).
*** In use.

OBMC = Outbound Midcourse Correction
IBMC = Inbound Midcourse Correction

Figure 4.1-9:B (Continued)

Fig. 4.1-9:A

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6) In the OVAM and (S+C), the PM-2 on either vehicle is backed up by the PM-3 on that vehicle (for orbit insertion), although use of such backup eliminates modular redundancy from "Mars Depart" on.

7) Failure of any module prior to scheduled use is assumed to be 100% detectable without error. Note that, as far as the logic of the model is concerned, it is not important when the PM-OBMC's and PM-IBMC's fail.

8) The EEM is only backed up (given availability of the backup) for failure prior to scheduled use.

The probability of successfully returning the crew to Earth was calculated for both the OVAM and single-vehicle mode, assuming that the mission was aborted at two different points in the mission profile: at successful Mars' orbit attainment and at successful Mars' landing. Definitions of conditional events pertinent to the OVAM analysis are shown in Table 4.1-5, and estimates of the probabilities of these events are calculated in Table 4.1-6. Figures 4.1-10 and -11 are diagrams of the models and calculations used for the OVAM analysis. The models used for analysis of the single-vehicle mode is represented by the first abort path in Figures 4.1-10 and -11. One means of interpreting the diagrams for the single-mode mission is to set P(A') = 1, P(B') = P(C') = 0, and all B vehicle module reliabilities equal to zero.

Table 4.1-5: OVAM CONDITIONAL EVENT DEFINITION

A = D = E = F = Event that no module failure has occurred up to time of abort (both vehicles).

B = Event that the only module failure occurring prior to abort is that of PM-2B.

C = Event that the only module failure occurring prior to abort is that of PM-2A.

Then: $P(A) = P[(\text{PM-1A}) \,\Omega\, (\text{PM-1B}) \,\Omega\, (\text{PM-OBMC-A}) \,\Omega\, (\text{MM-A}) \,\Omega\, (\text{MEM-B}) \,\Omega\, (\text{PM-2A}) \,\Omega\, (\text{PM-2B})]^*$

$$P(B) = \frac{P(A) \cdot P(\text{PM-3B}) \cdot Q(\text{PM-2B})}{P(\text{PM-2B})}$$

$$P(C) = \frac{P(A) \cdot P(\text{PM-3A}) \cdot Q(\text{PM-2A}) \cdot P(\text{Transfer})}{P(\text{PM-2A})}$$

A'= D'= E'= F'= Event that abort sequence is started under conditions such that event A is true.

B' = Event that abort sequence is started under conditions such that event B is true.

C' = Event that abort sequence is started under conditions such that event C is true.

Then: $P(A') = P(A)/(P(A) + P(B) + P(C))$

$P(B') = P(B)/(P(A) + P(B) + (P(C))$

$P(C') = P(C)/(P(A) + P(B) + P(C))$

*Up to time of abort.

Table 4.1-6: CALCULATION OF PROBABILITY OF CONDITIONAL EVENTS

$P(A) = (0.9949)\ (0.9949)\ (0.9985)\ (0.9985)\ (0.9510)\ (MEM)\ (0.9897)\ (0.9897)$

$P(B) = P(A)\ (0.9897)\ (0.0103)/(0.9897)$

$= P(A)\ (0.0103)$

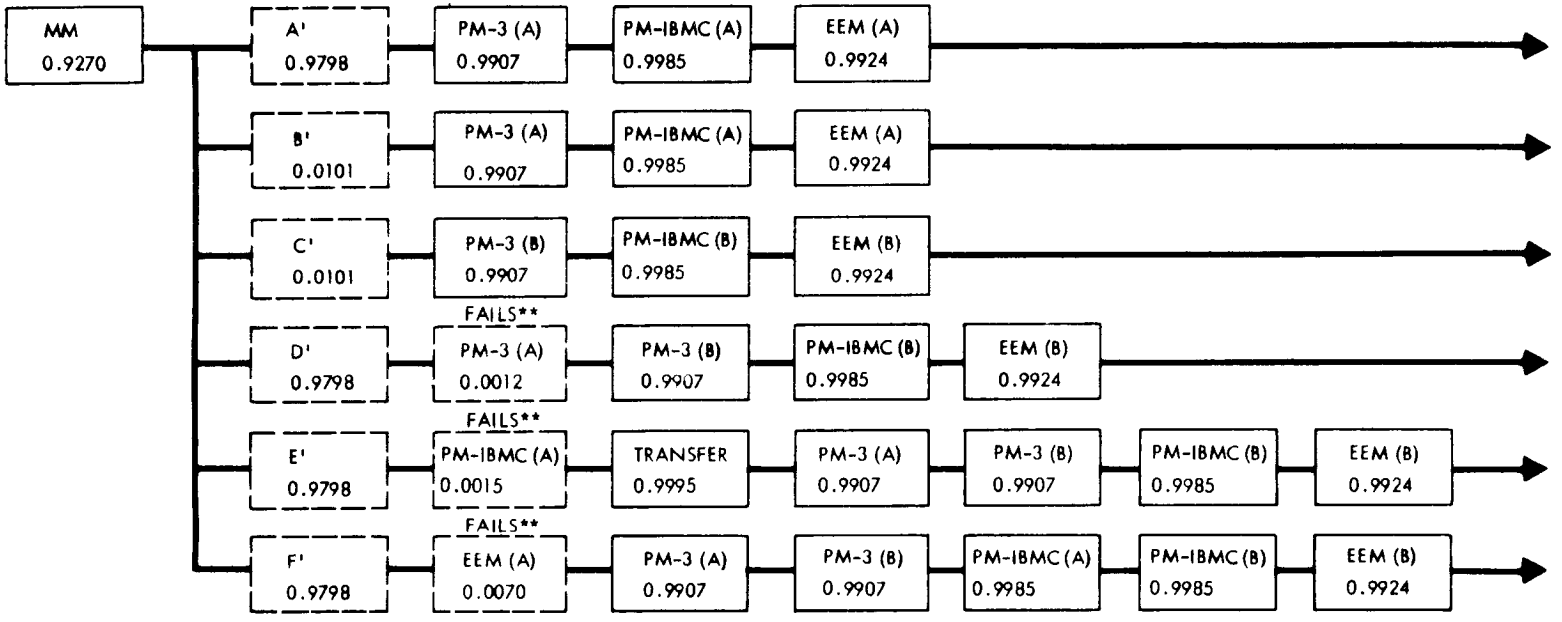
$P(C) = P(A)\ (0.9897)\ (0.0103)/(0.9897)$

$= P(A)\ (0.0103)$

$P(A') = 1/(1 + 0.0103 + 0.0103) = 1/(1.0206) = 0.97982$

$P(B') = 0.0103/(1.0206) = 0.01009$

$P(C') = 0.0103/(1.0206) = 0.01009$



OBMC = Outbound Midcourse Correction

IBMC = Inbound Midcourse Correction

* Ascent only.
** Before scheduled use.

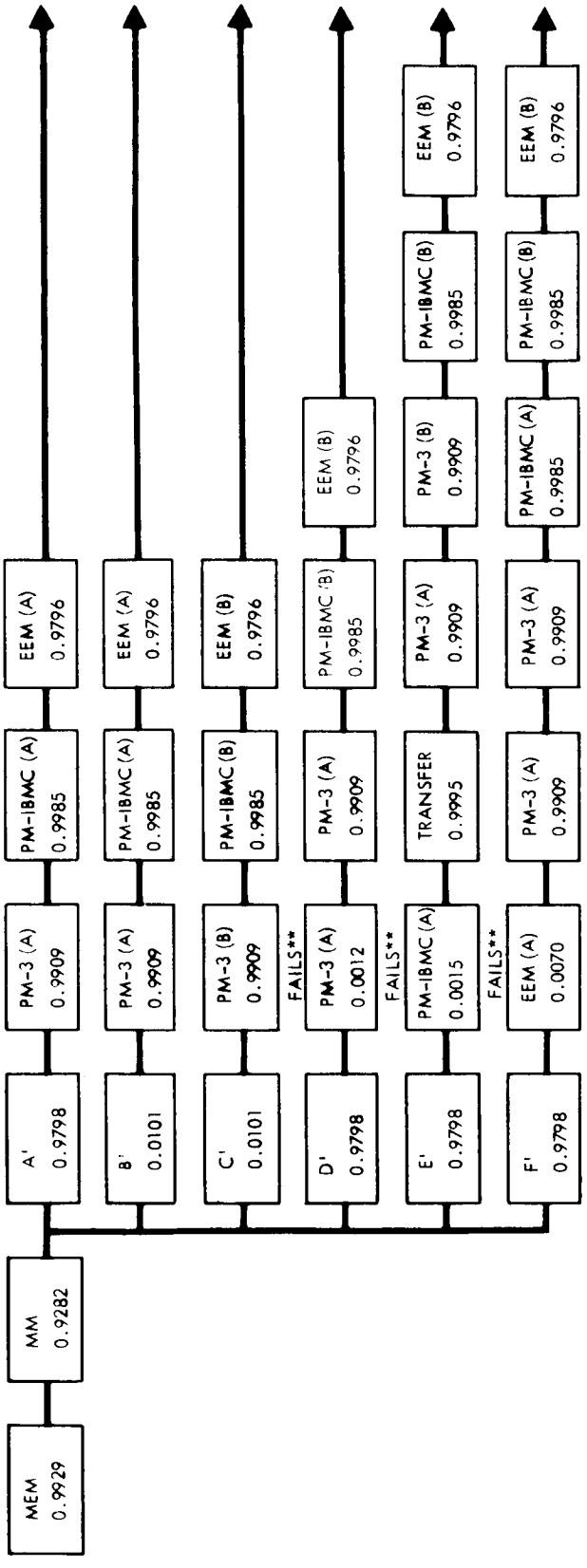
Unless otherwise noted, events described are from time of abort on.

$(1 + 0.00118 + 0.00146 + 0.00679)(0.98168) = 0.9909$

$0.9909\ (0.9270) = 0.9186$ (OVAM)

$0.98168\ (0.9270) = 0.9100$ Single

Figure 4.1-10: ABORT PATHS — FROM MARS ORBIT INSERTION (OVAM)



* Ascent only.
** Before scheduled use.
Unless otherwise noted, events described are from time of abort on.

OBMC = Outbound Midcourse Correction
IBMC = Inbound Midcourse Correction

$(1 + 0.00118 + 0.00146 + 0.00679)(0.96923) = (1.0094)(0.96923) = 0.9783$

$0.9783\ (0.9216) = 0.9016$ OVAM
$0.9216\ (0.96923) = 0.8932$ Single

Figure 4.1-11: ABORT PATHS — FROM MARS LANDING (OVAM)

4.2 ABORT CAPABILITY

A decision to abort the mission is essentially a decision to try to reduce the trip time and could arise from a number of possible situations. For instance, any damage or failure that results in leakage or loss of part of the expendables necessary to life support or midcourse maneuvers could give rise to an abort decision. For this reason, potential abort trajectories were calculated for each of six representative missions:

1982 Mars Opposition
1986 Mars Opposition
1986 Mars Conjunction
1982 Mars with Venus Swingby
1980 Venus with Short Staytime
1980 Venus with Long Staytime

The abort trajectories must be considered idealized from the standpoint that the actual flight profiles were not constrained as to perihelion or aphelion distances, but they do provide reasonable estimates of the abort requirements from any point on nominal outbound trajectories for varying return trip times. Figures 4.2-1 through 4.2-6 present matrices of abort ΔV requirements for each mission as a function of "time from Earth departure" and "time to return from abort". The matrices of abort impulse numbers are given in kilometers per second and represent only that impulse required to divert the vehicle's flight from its nominal outbound trajectory to a return trajectory of given return time and having an Earth entry velocity of 19,800 m/sec or less. Three abort modes are shown for each mission:

1) Using PM-2 and PM-3 in sequence

2) Using PM-3 only

3) Using PM-2 only, but with PM-1 fuel jettisoned.

The first of these is the desirable mode. However, if PM-2 is not available due to malfunction or damage, the second mode would be used. If PM-3 could not be used for any reason, then all of the propellant (PM-2 and PM-3) could be burned through PM-2. This case gives virtually identical results to Case 2 and is not shown by a separate line. The third case might result if fuel could not be transferred from PM-3 to PM-2 but could be dumped when PM-3 is unusable. In all modes shown, the unnecessary weights, MEM, and/or probes and associated interstages are jettisoned.

All of the abort possibilities for a particular PM or combination of available PM's are shown enclosed either above, to the right, or within the applicable contour lines. Proceeding from left to right on the charts, the first contour line shown represents the minimum return trip time capability of the particular vehicle. In the Mars 1982 opposition mission of Figure 4.2-1, if the vehicle is all-propulsive and can use both PM-2 and PM-3 stages for the abort maneuver, the longest of the minimum abort return trip time requirements would occur at a point about 145 days after Earth departure. This would necessitate a return trip time of about 200 days. If both PM-2 and PM-3 can be used, the vehicle

would have abort capabilities throughout the entire outbound leg. The aerobraker, which would have only the PM-3 stage available for abort, would also have full abort capability for this mission, and the longest of its minimum return trip times would be about 235 days. Longer return trip trajectories are available in either case, if a portion of the ΔV of the stage cannot be acquired. If the all-propulsive vehicle has only PM-2 available for abort, the regions of abort are severely limited, covering a period of the outbound leg from 10 to 55 days after Earth departure.

Table 4.2-1 presents tabulated ranges of abort possibilities during the outbound leg of each of the six representative missions. The all-propulsive vehicles with both PM-2 and PM-3 available to perform the abort maneuver can abort any time along the outbound legs of each mission. The aerobraker or all-propulsive vehicle can do likewise with only PM-3 available except for the 1986 Mars conjunction mission where only 50 days of abort capability is available.

If only PM-2 is available for abort in the all-propulsive vehicle, the abort capabilities are again severely limited. Other abort possibilities should be investigated in subsequent studies, including out-of-sequence usage of propulsive stages. However, the design implications of providing such a capability should also be considered.

Table 4.2-1: MISSION OUTBOUND LEG ABORT POSSIBILITIES

	Range of Abort Capabilities (Days from Earth Departure)		
Mission Designation	$\Delta V_2 + \Delta V_3$ Available (All-Propulsive)	ΔV_2 Only Available (All-Propulsive)	ΔV_3 Only Available (Aerobraker or All-Propulsive)
Mars 1982 Opposition	Full Leg 5-285	10-55	Full Leg 5-285
Mars 1986 Opposition	Full Leg 5-185	5-35	Full Leg 5-185
Mars 1986 Conjunction	Full Leg 5-205	15-35	Full Leg 5-205
Mars 1982 with Venus Swingby	Full Leg 5-255	5-45	Full Leg 5-255
Venus 1982 with Short Stay	Full Leg 5-125	10-20	Full Leg 5-125
Venus 1982 with Long Time	Full Leg 5-185	20-30 85-185	Full Leg 5-185

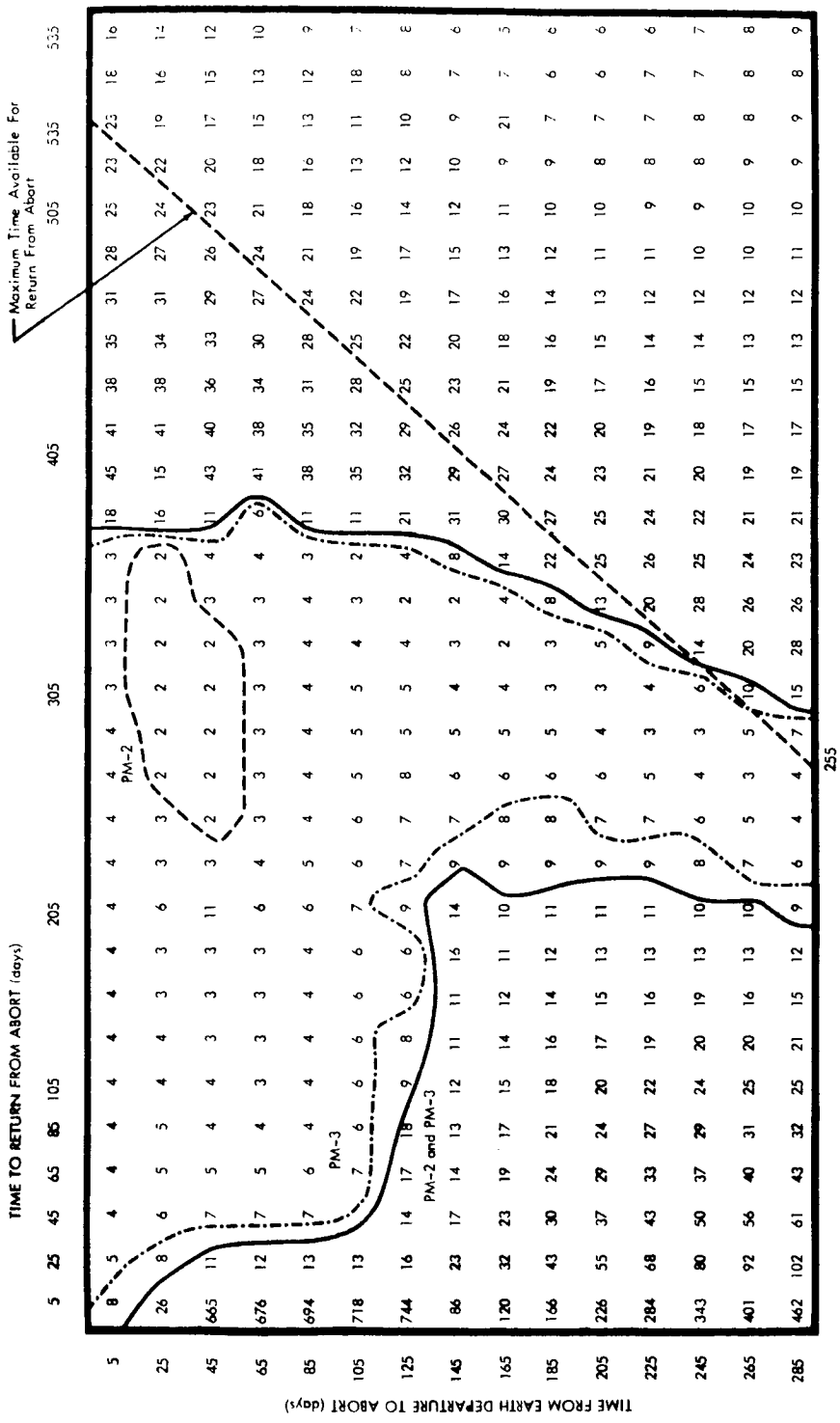


Figure 4.2-1: MARS 1982 OPPOSITION MISSION ABORT CAPABILITY (540-Day Mission) Abort Velocity (km/sec)

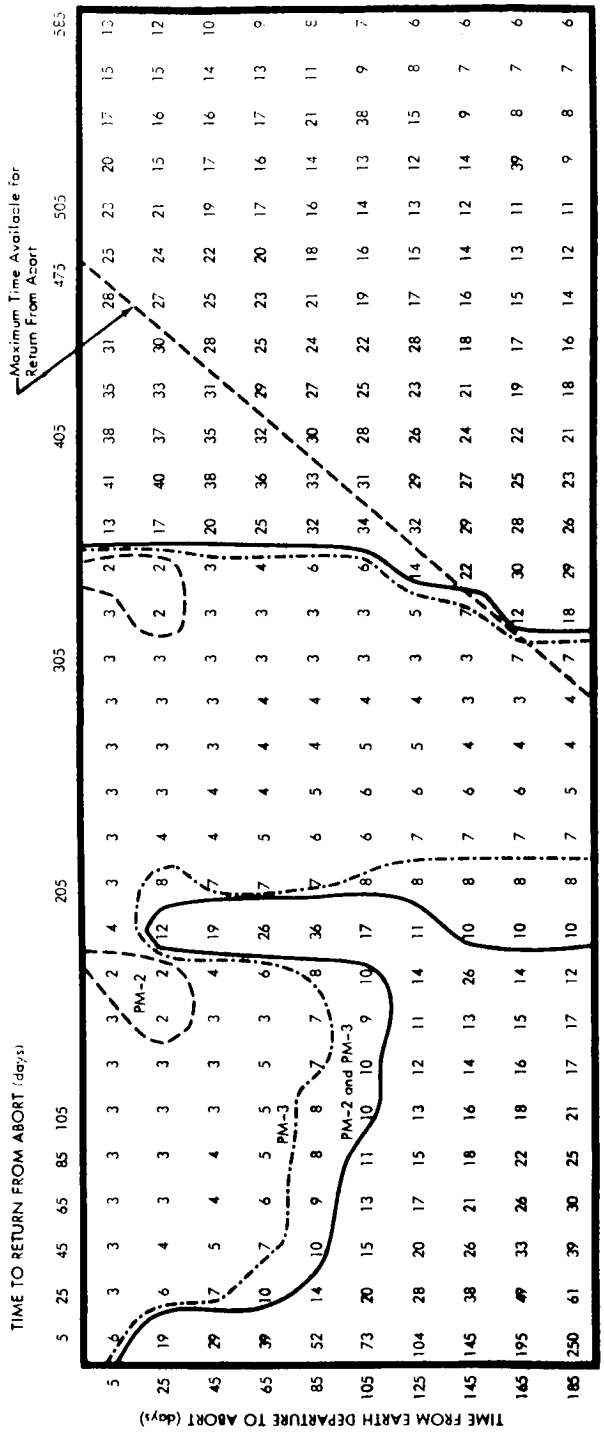


Figure 4.2-2: MARS 1986 OPPOSITION MISSION ABORT CAPABILITY
(480-Day Mission) Abort Velocity (km/sec)

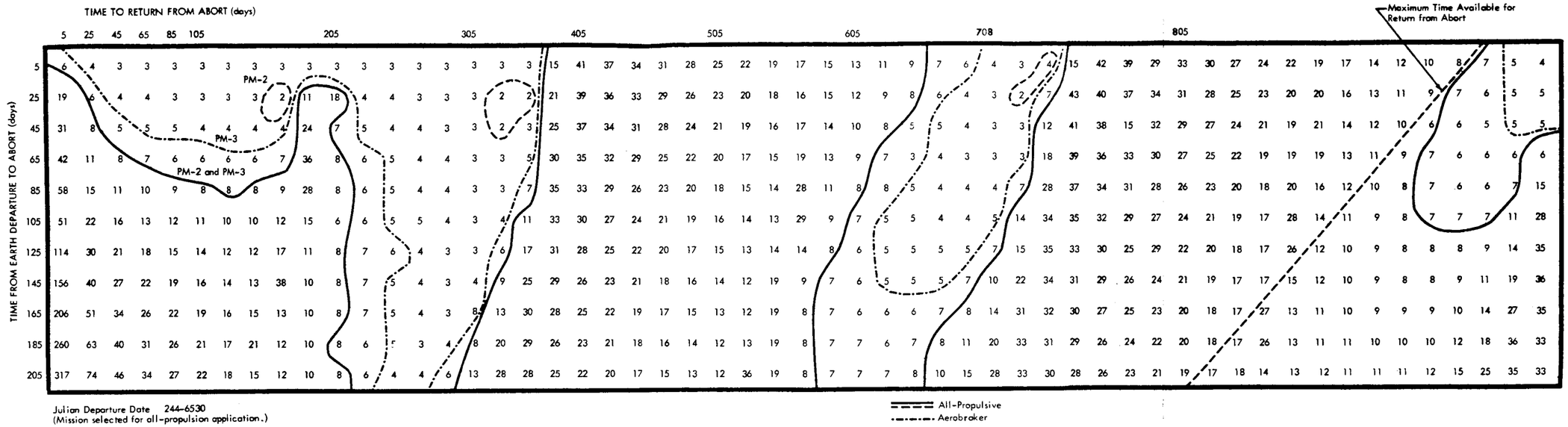


Figure 4.2-3: MARS 1986 CONJUNCTION MISSION ABORT CAPABILITY (1040-day Mission)
Abort Velocity (km/sec)

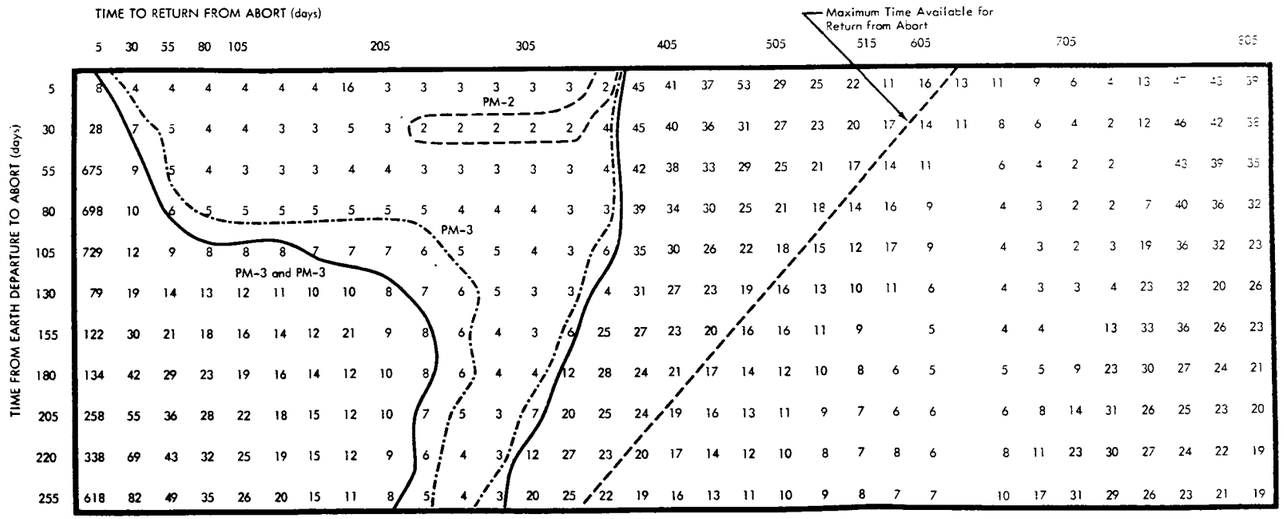


Figure 4.2-4: A MARS 1982/VENUS SWINGBY MISSION ABORT CAPABILITY (600-day Mission)
Abort Velocity (km/sec)

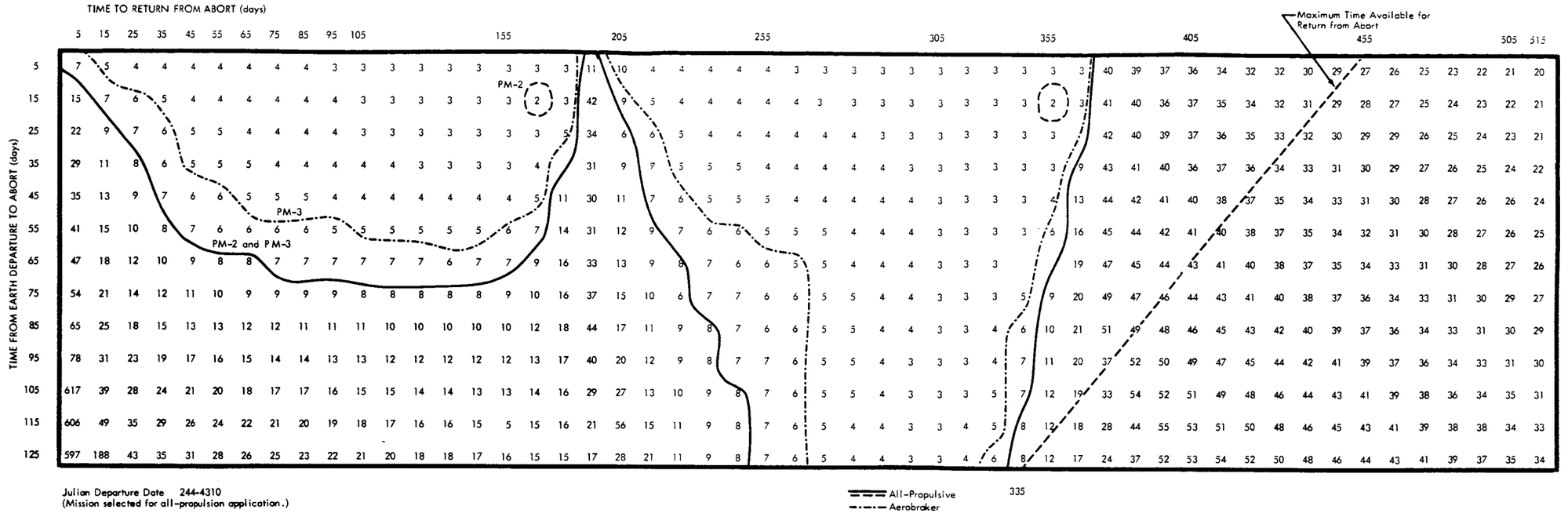
Fig 4.2-4: B

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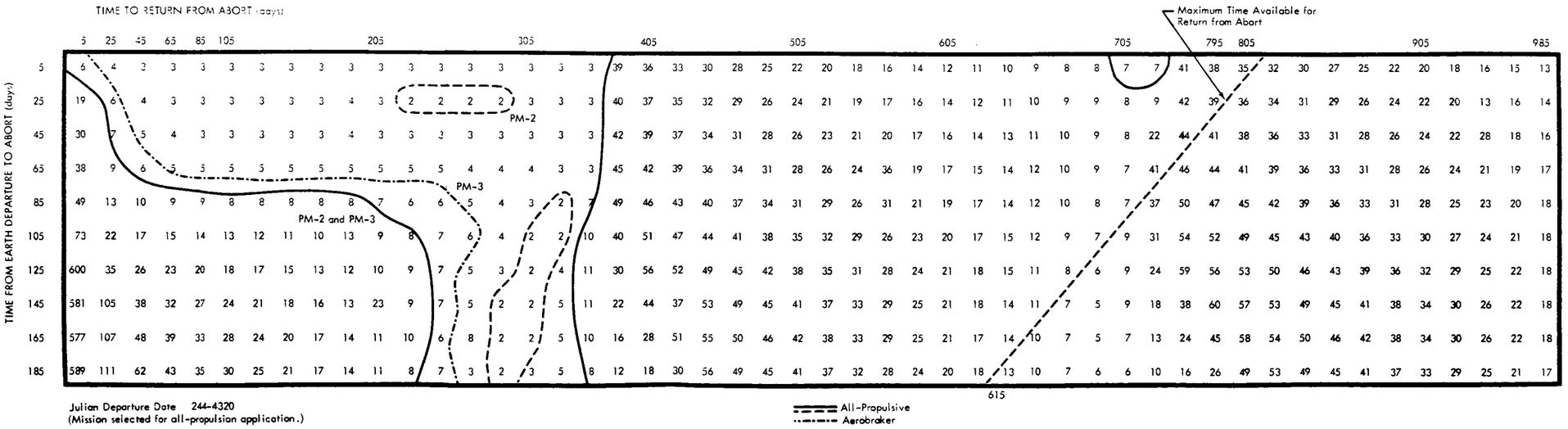
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Julian Departure Date 244-4310
(Mission selected for all-propulsion application.)

Figure 4. 2-5: VENUS 1980 / W SHORT STOPOVER MISSION ABORT CAPABILITY
(460-day Mission) Abort Velocity (km/sec)



Julian Departure Date 244-4320
(Mission selected for all-propulsion application.)

Figure 4. 2-6: VENUS 1980 / W LONG STOPOVER MISSION ABORT CAPABILITY
(800-day Mission) Abort Velocity (km/sec)

Fig. 4-2-6: B

229

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In each of the abort requirement charts (Figures 4.2-1 through 4.2-6), a broken line is drawn diagonally across the chart in a position representing the maximum time available for return from abort. This time assumes that the space vehicle is designed for the nominal mission time (indicated above the chart in the middle of the page). Therefore, the maximum abort return trip time is calculated from "total mission time minus time from Earth departure to abort". In none of the six missions are the abort capabilities limited by this time constraint.

Another important item that became evident from the abort capability contours is the fact that there are relatively short periods during the outbound trips during which the "return-from-abort" trip time increases rapidly. Figure 4.2-1 shows that if an abort is made at about 105 days after Earth departure, the return trip could take as little as 45 days, whereas 25 to 30 days later, the return trip would take at least 195 days - over four times as long. This is typical for all of the six missions investigated.

By combining the reliability estimates of Section 4.1 with the results of the abort trajectory analysis, estimates were made of the probability of successful return from various points along the outbound leg and at the planet. The results for a 1986 Mars opposition mission of 480 days duration are shown in Figure 4.2-7. In that figure, an assumption is made, based on the lower of the two sets of reliability estimates of Section 4.1, that both PM-2 and PM-3 are used. The rapid increase in return trip time noted previously occurred at about 65 days for this mission and accounts for the sudden drop in return probability to the level that exists at the planet.

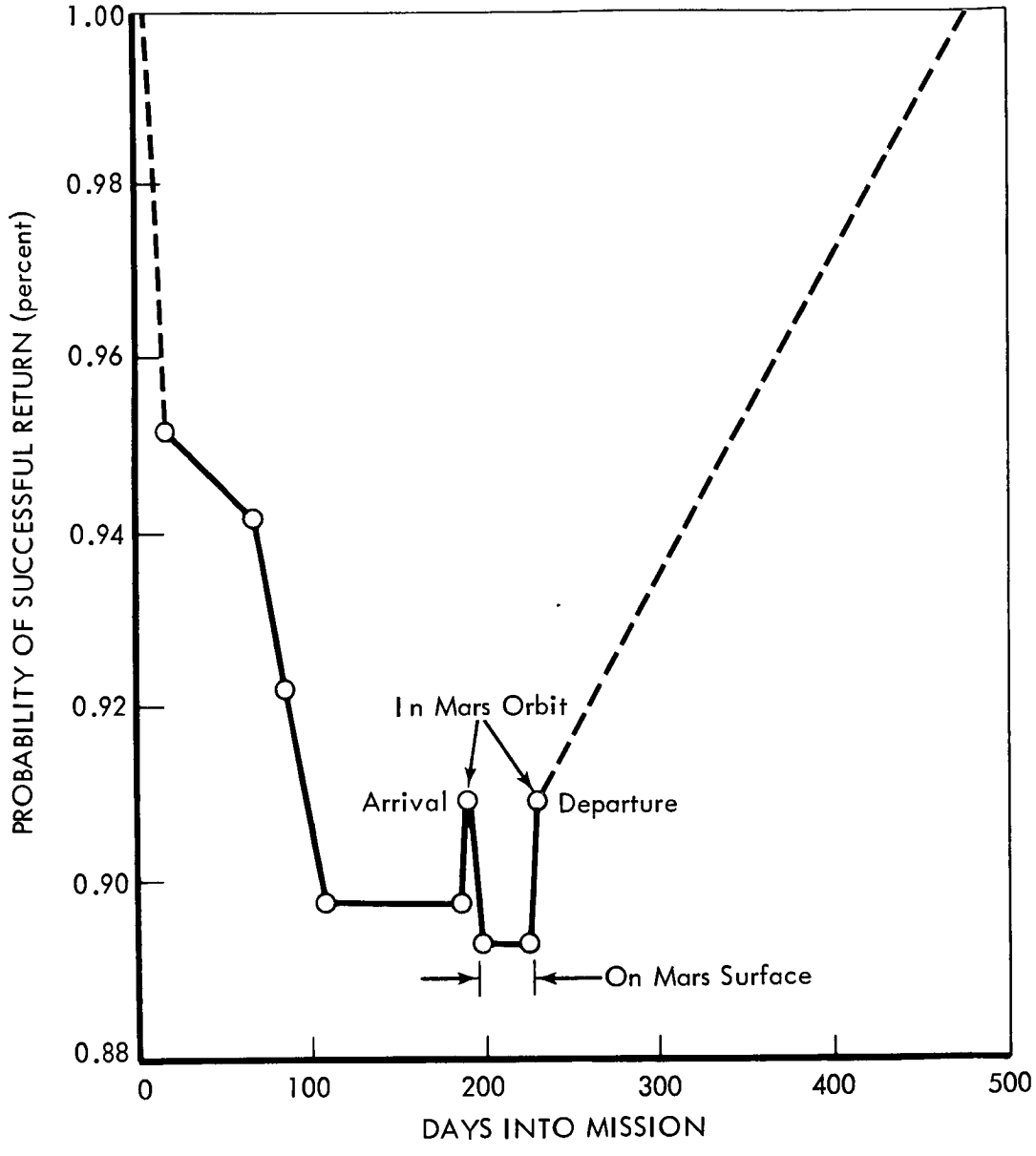


Figure 4.2-7: PROBABILITY OF SUCCESSFUL RETURN

5.0 MISSION ENVIRONMENTS

The environments used in the space vehicle design process are presented in this section. These include the Earth surface, Earth orbital, interplanetary, Mars, and Venus environments. The latter three, at best, are not very well known. For this study, nominal environments were selected for design purposes, and maximum and minimum environments were chosen for use in determining the sensitivities of the space vehicle design to the uncertainties in the environments. With the exception of the meteroid, corpuscular radiation, and planet atmospheres, the environments selected for use in the IMISCD study were obtained from Boeing Document D2-82746-1*. The Mars model atmospheres VM-3, VM-7, and VM-8 were selected by NASA for use in this study. Boeing designated Mars model atmosphere VM-7 for use in the design studies and VM-3 and VM-8 for use in space vehicle sensitivity studies.

5.1 EARTH SURFACE ENVIRONMENT

The surface environmental characteristics were chosen to represent the climatic extremes of the Eastern Test Range (ETR).

5.1.1 TEMPERATURE

Based on records of the last 50 years, the ETR extremes in air temperature are 37°C (99°F) and -2°C (28°F).

5.1.2 AMBIENT PRESSURE

Ambient pressure extremes for ETR are 1041 mb (30.6 in. Hg) to 900 mb (26.6 in. Hg).

5.1.3 HUMIDITY

An extreme relative humidity between 75 and 100% and an air temperature between 22.8°C (73°F) and 27.8°C (82°F) can be expected for a period of 15 days. A relative humidity of 100% may occur one-fourth of the time at the lower temperature in cycles not exceeding 24 hours.

5.1.4 SOLAR RADIATION

The maximum expected solar intensity is 120 w/ft^2 with constituents as follows:

Ultraviolet	6%
Visible	44%
Infrared	50%

*Boeing Document D2-82746-1, *Flight Spacecraft Environmental Criteria—Voyager*, December 1966

5.1.5 ATMOSPHERE

The atmosphere used is U.S. Standard Atmosphere - 1962.

5.2 EARTH ORBITAL ENVIRONMENT

5.2.1 AMBIENT PRESSURE

The ambient pressure at the initially assumed 289-nautical-miles (535 km) assembly orbit is 3.3×10^{-10} in. of Hg.

5.2.2 TERRESTRIAL INFRARED RADIATION

Using an average Earth temperature of -20°C and an average emittance of 1, the radiation to space is 21 w/ft^2.

5.2.3 SOLAR RADIATION

The solar intensity is 130 w/ft^2 with constituents as follows:

Ultraviolet	7.3%
Visible	47.7%
Infrared	45.0%

During passage through the penumbra into total shadow, the variation of solar intensity may be approximated by a linear function of time.

5.2.4 REFLECTED SOLAR RADIATION

With an average terrestrial albedo of 0.36, the reflected radiation is 45.2 w/ft^2.

5.2.5 CHARGED PARTICLE RADIATION

The nominal radiation environment with the fluxes and spectra of the significant components for low Earth orbit and passage through the radiation belts are listed below.

5.2.5.1 Low Earth Orbit

Protons ($J > E$) (No. $>$ E/cm^2-day)

$5 \times 10^7 \exp(-E/6.5)$	$4 \leq E \leq 15$
$6 \times 10^6 \exp(-E/83)$	$15 \leq E \leq 30$
$7.5 \times 10^6 \exp(-E/65)$	$30 \leq E \leq 50$
$6 \times 10^6 \exp(-E/70)$	$50 \leq E$

Electrons ($E >$ E/cm^2-day)

$$J(> E) = 3 \times 10^{10} \exp(-E/0.75)$$

$$J(> E) = 6 \times 10^{10} \exp(-E/0.14) + 1.5 \times 10^7 \exp(-E/4)\, e^{-\frac{(Y-1968)}{\tau}}$$

where Y is the year and τ lies between 1 and 1.5 years.

Galactic Cosmic Rays

Use 50% of interplanetary (see Section 5.3.3.2) above the Stormer cutoff, conservatively approximated by $P = 1.5 \times 10^4 \cos^4 i$, where i is the orbit inclination angle.

The nominal trapped particle environment is given for a randomly initiated 290-nautical-miles altitude, 30 degree inclination orbit. The mission calls for an orbit that executes precisely 15 orbits per day and has the same ground track day after day. A region of intense trapped flux is permanently located over what is called the south Atlantic anomaly. In this region, due to geophysical processes, the Earth's magnetic field is distorted in such a fashion as to introduce large particle fluxes at relatively low altitudes. Because the location of the spacecraft ground track across this region will remain fixed, the exact Earth orbit will exert a large influence on the received flux.

A pictorial description of the trapped flux in the region of the anomaly is shown in Figure 5.2-1 for a slightly higher altitude (600 km). Contours of the flux of protons of energy greater than 15 Mev and of electrons of energy greater than 0.5 Mev are shown. These particles are sufficiently energetic to penetrate a typical space suit. The geographical region of encountered trapped radiation will be similar at 289 nautical miles (535 km) but reduced in size and magnitude.

5.2.5.2 Belt Passage $(No./> E)/cm^2$)

Typical low level passage gives:

Electrons

$$J(> E) = 3 \times 10^{10} \exp(-E/0.25) + 1.1 \times 10^{10} \exp(-E/0.45)$$

$$J(> E) = 1.25 \times 10^{9} \exp(-E/2.5) + 1.1 \times 10^{8} \exp(-E/8) + 1.5 \times 10^{7} \exp(-E/12.5)$$

$$J(> E) = 1.25 \times 10^{7} \exp(-E/5) + 2.5 \times 10^{6} \exp(-E/12.5)$$

For maximum level passage multiply by 10.

5.2.6 METEROID FLUX

The following flux equations were used for the ranges shown:

$$\log N = -1.34 \log m - 13.44 \qquad m > 10^{-1.3} \text{ gm}$$

$$\log N = -\log m - 13.00 - A \log R \qquad 10^{-1.3} > m > 10^{-7} \text{ gm}$$

$$\log N = -0.64 \log m - 10.44 - A \log R \qquad 10^{-7} > m > 10^{-10} \text{ gm}$$

where $A = 1.5$

R = distance from Sun in A.U.

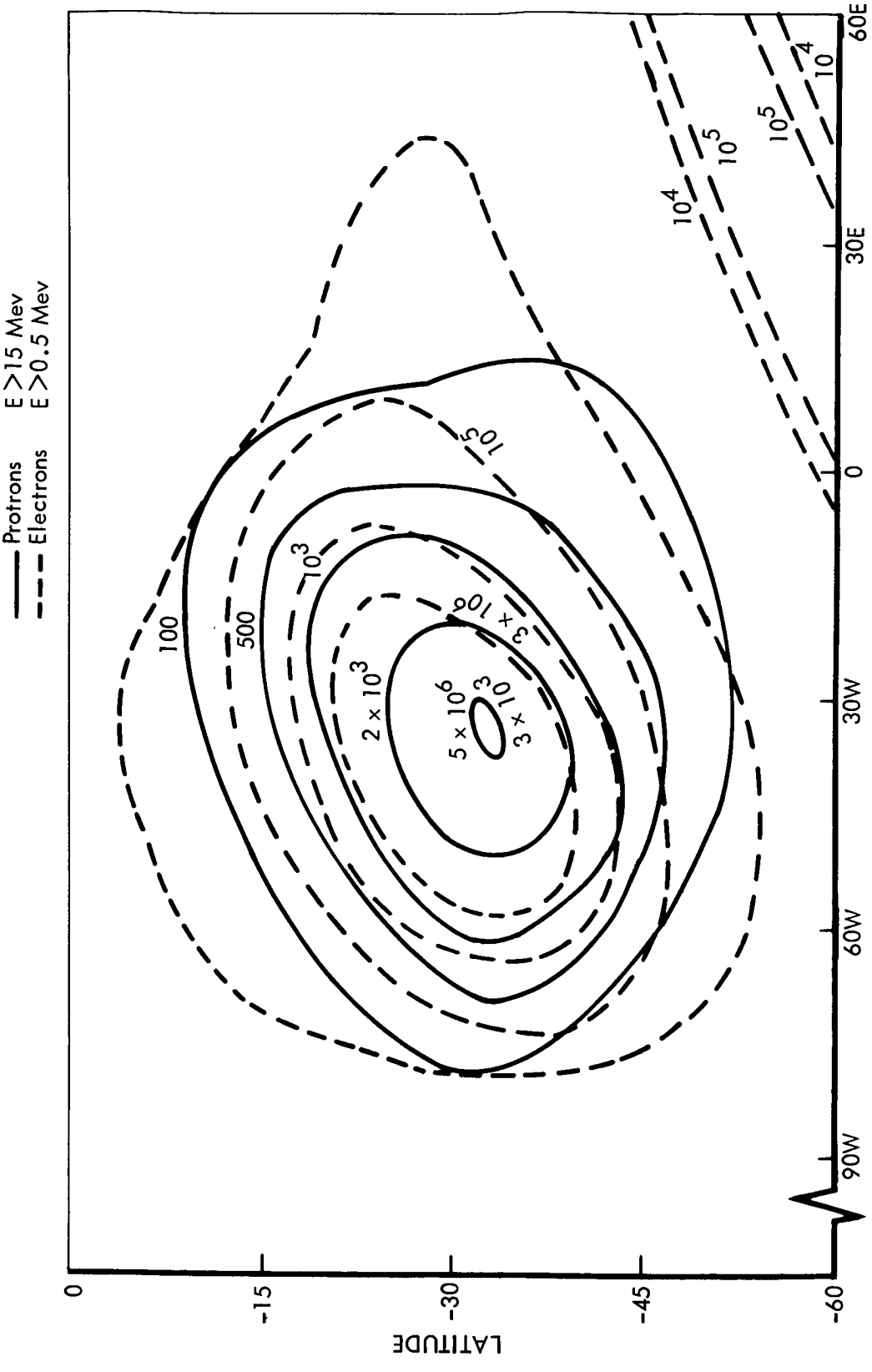


Figure 5.2-1: TRAPPED PARTICLE FLUX AT LOW ALTITUDE — 600 KM

These fluxes have been plotted in Figure 5.2-2 for near Earth. By empirical fit, the recommended particle density value that can be used for the entire range is:

$$\rho = 0.25\ m^{-1/8}\ gm/cm^3$$

5.3 INTERPLANETARY ENVIRONMENT

5.3.1 AMBIENT PRESSURE

The ambient pressure is that of space and is taken to be 10^{-14} in. Hg.

5.3.2 SOLAR RADIATION

The solar intensity variation is shown in Figure 5.3.1. The constituents are as follows:

Ultraviolet	7.3%
Visible	47.7%
Infrared	45.0%

5.3.3 CHARGED PARTICLE RADIATION

The nominal radiation environment with fluxes and spectra of significant components for the interplanetary flight phase are shown below.

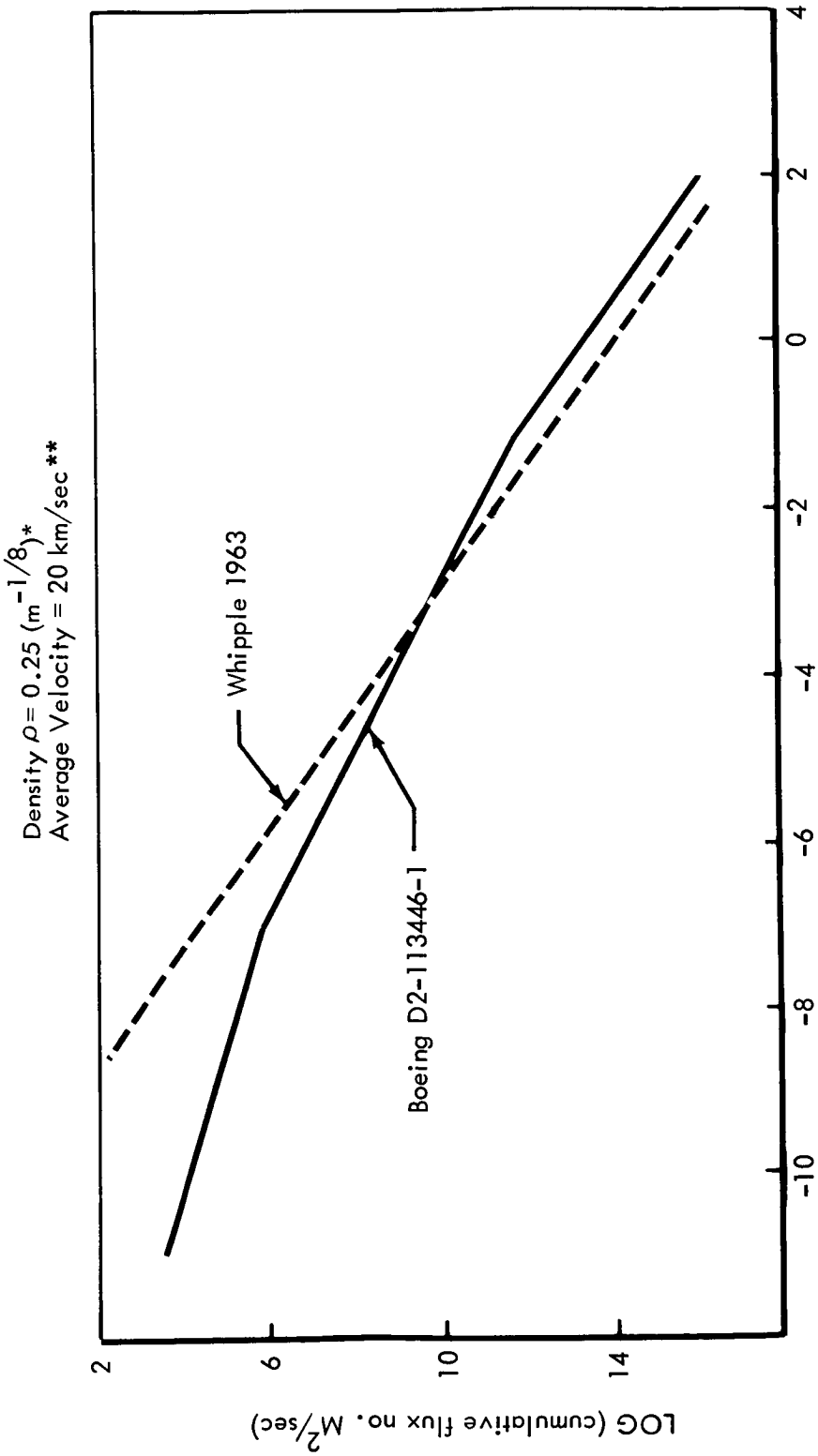
5.3.3.1 Solar Particle Radiation

		Protons	Alphas
Solar Wind	Flux (cm^{-2} sec)	$2x10^2$	$1\text{-}2x10^7$
	Temp (°K)	$2x10^5$	$8x10^5$
	Velocity (km/sec)	400-600	400-600
Storm Particles	Flux (cm^{-2} sec)	$2x10^9$	$1\text{-}2x10^8$
	Temp (°K)	$8x10^5$	$3x10^6$
	Velocity (Kev)	800-1000	800-1000

Solar Particle Events (No. > P/cm^2-year)

$$J_{protons}\ (>P) = 6x10^8\ \exp(-P/22) + 5.4x10^7\ \exp(-P/73)\ \text{(solar minimum)}$$
$$1.3x10^{12}\ \exp(-P/24) + 1.2x10^{10}\ \exp(-P/113)\ \text{(solar maximum)}$$

where $P = \sqrt{E^2 + 1876\ E}$



Reference: *Verniani **NASA TN D-2958 & Doynanyi

Figure 5.2-2: NEAR EARTH METEOROID ENVIRONMENT

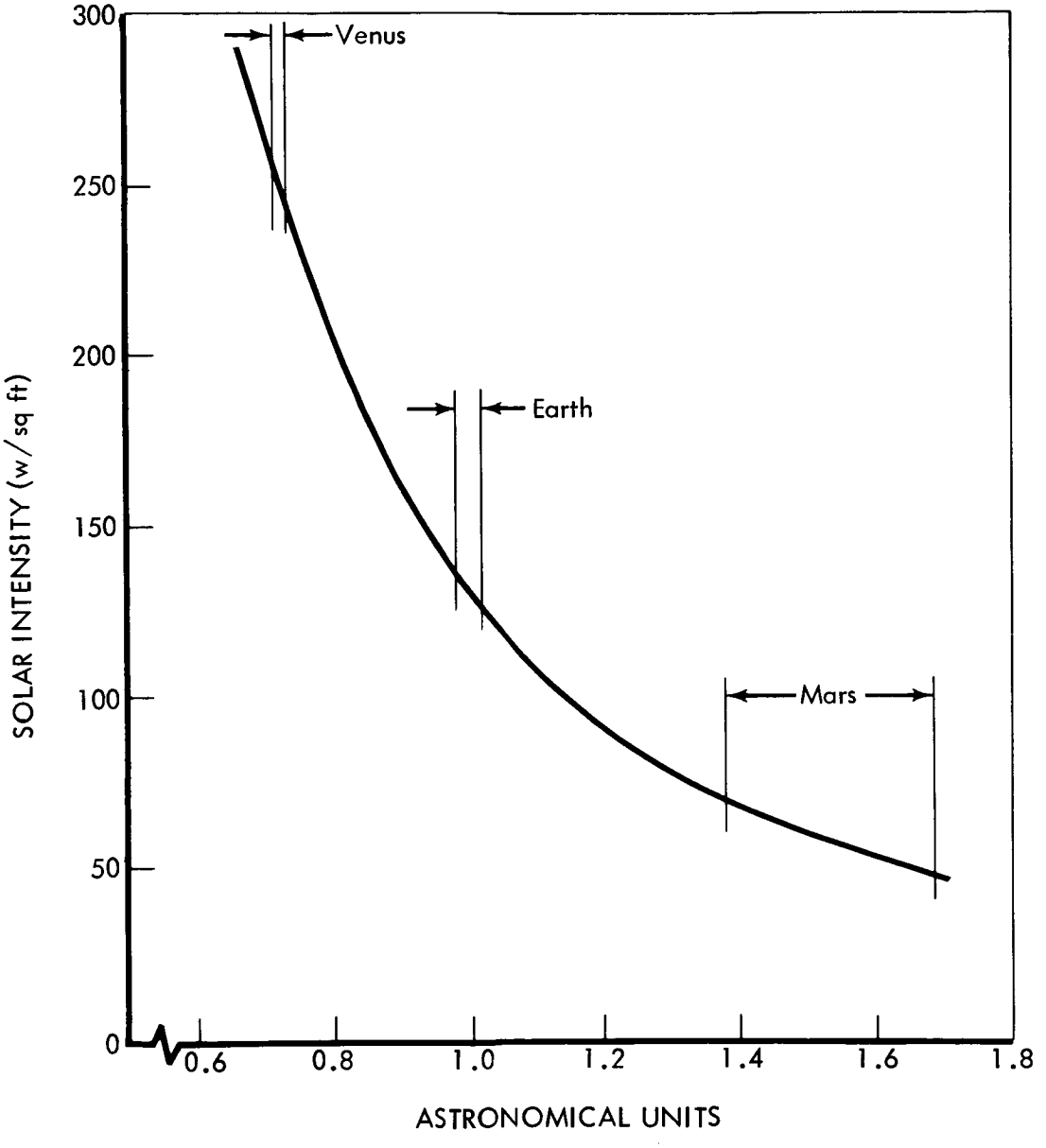


Figure 5.3-1: SOLAR INTENSITY

Charge composition of solar events particles

Proton/Alpha	Variable 100:1 to 1:1
Alpha/Medium ($E \leq Z \leq 9$)	60:1
Light ($3 \leq Z \leq 5$)/Medium	<0.02
Heavy ($Z \geq 10$)/Medium	0.1
Deuteron/Proton	$\cong 10^{-3}$
Electron/Proton	Small-maybe variable

The above data represent the nominal environment at 1 A.U. The solar wind and storm environment was based largely on Mariner 2 results*, (1966). The solar wind flux on the average is $\sim 2 \times 10^{-8}$ cm $^{-2}$sec^{-1}. On occasion the flux, bulk velocity, and temperature increase and then decrease to normal in a few hours.

The Mariner and Pioneer data are consistent with an inverse R^2 dependence of the of the flux of the solar plasma. We propose this for all the solar particles, although there is less substantiating data that the R^{-2} relation holds for the solar particle events. This variation for the events is dwarfed by that associated with the solar cycle. The solar particle event spectra including both the solar cosmic rays and the energetic storm particles are given for solar maximum and solar minimum. These resulted from correlations of events to sunspot number for cycle 19 (1954-1964). The spectra in the data above are less severe than that obtained near the maximum of cycle 19- which is:

$$J(>P) = 8.6 \times 10^{12} \exp(-P/21) + 5 \times 10^{10} \exp(-P/94) \text{ No.}(>P)/\text{cm}^2 \text{ yr}$$

However, this severe an environment is not expected during the next two cycles.

Also, a model event is not given in the data above. By adding the two largest events of cycle 19, February 13, 1956 and November 12, 1960, a reasonably large model event could be generated. To estimate the importance of the model event, we give here the integrated spectrum, J(>P) for the February 13, 1956 and November 12, 1960 events:

$$J(>P) = 9 \times 10^{11} \exp(-P/22) + 1.1 \times 10^{10} \exp(-P/155) \text{ No.}(>P)/\text{cm}^2$$

*Marcia Neugebauer and Conway Snyder, "Mariner 2 Observations of the Solar Wind," *Journal of Geophysical Research*, Vol 71, October 1966, p. 4469

It is clear that the hardness of this model spectrum compared to the nominal event spectrum given in the summary will affect the conclusions of any shielding study.

5.3.3.2 Galactic Cosmic Radiation

The differential spectrum of protons and alphas for sunspot minimum are portrayed in Figure 5.3-2. Also shown is the proton spectrum for solar maximum.

The variation in flux from Venus to Mars is expected to be slight compared to the solar cycle variation. A factor of two increase in the spectrum under 1 Bev at solar minimum is sufficiently conservative for the spectrum at Mars.

Charge Composition of Galactic Cosmic Radiation

Element	Atomic No., Z	Percent of Total
Hydrogen	1	80 to 85
Helium	2	11 to 16
Light Nuclei	$3 \leq Z \leq 5$	2
Medium Nuclei	$6 \leq Z \leq 9$	1
Heavy Nuclei	$Z \geq 10$	3

The galactic cosmic rays are considered to be most significant for extended manned missions since they are most difficult to shield against. In fact, the hazard increases with shielding due to production of secondary and induced activity in the shield material. Thus, this environment is covered in more detail than is usual with shorter term or unmanned missions. Shielding studies may show that even more detail is required by the study.

5.3.4 METEOROID

The interplanetary meteoroid environment is the same as described in Section 5.2.6.

5.4 MARS ENVIRONMENT

5.4.1 MODEL ATMOSPHERES (NASA)

Mars model atmosphere VM-8 will be used for space vehicle design. Mars model atmospheres VM-3 and VM-7 will be used for sensitivity studies of the effects of atmosphere uncertainties on space vehicle design. These model atmospheres were constructed from data in Table 5.4-1, and are described in Tables 5.4-2, -3, and -4.

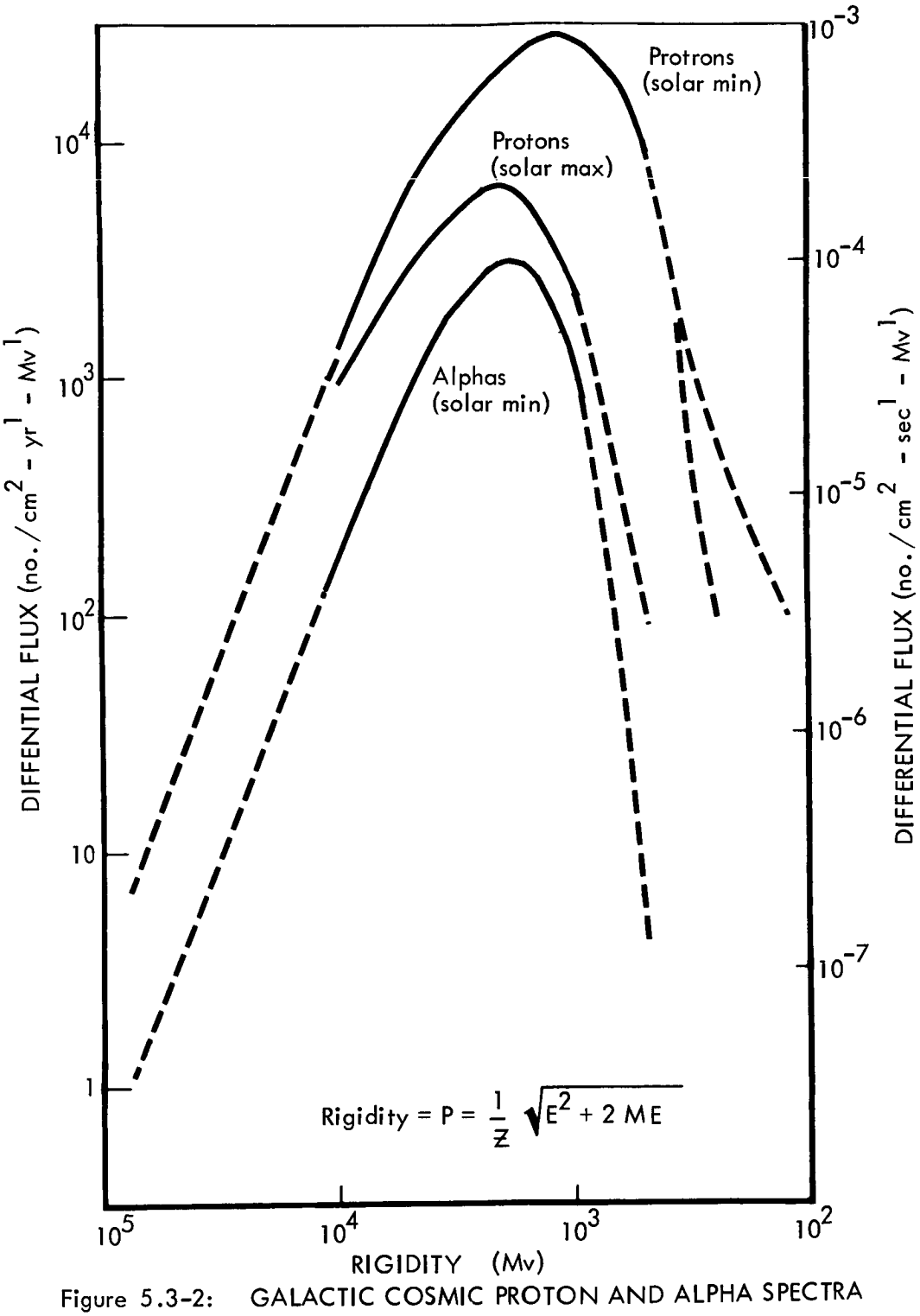


Figure 5.3-2: GALACTIC COSMIC PROTON AND ALPHA SPECTRA

Table 5.4-1: VOYAGER/MARS ENGINEERING MODEL ATMOSPHERES

Property	Symbol	Dimension	VM-3	VM-8	VM-7*
Surface Pressure	P_o	mb.	10.0	5.0	5.0
		lb ft ^{2}	20.9	10.4	10.4
Surface Density	ρ_o	(gm. cm^{3})10^{5}	1.365	1.32	0.68
		(slugs ft ^{3})10^{5}	2.65	2.56	1.32
Surface Temperature	T_o	°K	275	200	275
		°R	495	360	495
Stratospheric Temperature	T_R	°K	200	100	200
		°R	360	180	360
Acceleration of Gravity of Surface	G	cm/sec.^{2}	375	375	375
		fps^{2}	12.3	12.3	12.3
Composition					
CO_2 (by mass)			28.2	100.0	28.2
CO_2 (by volume)			20.0	100.0	20.0
N_2 (by mass)			71.8	0.0	71.8
N_2 (by volume)			80.0	0.0	80.0
A (by mass)			0.0	0.0	0.0
A (by volume)			0.0	0.0	0.0
Molecular Weight	M	mol^{-1}	31.2	44.0	31.2
Specific Heat of Mixture	C_P	cal/gm. C	0.230	0.166	0.230
Specific Heat Ratio	γ		1.38	1.37	1.38
Adiabatic Lapse Rate	Γ	°K/km	-3.88	-5.39	-3.88
		°R/1000 ft	-2.13	-2.96	-2.13
Tropopause Altitude	h_T	km	19.3	18.6	19.3
		1000 ft	63.3	61.0	63.3
Inverse Scale Height		km^{-1}	0.0705	0.199	0.0705
(stratosphere)		ft^{-1}x10^{5}	2.15	6.07	2.15
Continuous Surface Wind Speed	V	fps	155.5	220.0	220.0
Peak Surface Wind Speed	v_{max}	fps	390.0	556.0	556.0
Design Vertical Wind Gradient		fps/1000 ft	2	2	2

*VM=Voyager model. Dash numbers indicate consecutive models developed for design purposes.

Table 5.4-2: JPL MARS ATMOSPHERE, VM-3

Altitude Geopotential (km)	Altitude Geometric (km)	Temperature (°K)	Pressure (mb)	Density (gm/cm^3)	Density Scale Height (km)
0	0	275	10.0	1.36×10^{-5}	27.0
2	2.00	267	9.01	1.26	26.2
4	4.00	259	8.10	1.17	25.4
6	6.01	252	7.26	1.08×10^{-5}	24.7
8	8.02	244	6.48	9.96×10^{-6}	23.9
10	10.03	236	5.76	9.15	23.2
12	12.04	228	5.10	8.38	22.4
14	14.06	221	4.50	7.66	21.6
16	16.08	213	3.95	6.97	20.9
18	18.10	205	3.46	6.32	20.1
19.2	19.31	200	3.18	5.95	19.7
20	20.12	200	3.00	5.62	14.2
24	24.17	200	2.27	4.25	14.2
28	28.23	200	1.71	3.21	14.2
32	32.30	200	1.29	2.42	14.2
36	36.38	200	9.78×10^{-1}	1.83	14.2
40	40.48	200	7.38	1.38	14.2
44	44.58	200	5.58	1.04×10^{-6}	14.2
48	48.69	200	4.21	7.88×10^{-7}	14.2
50	50.75	200	3.66	6.85	14.2
60	61.08	200	1.81×10^{-1}	3.40	14.2
70	71.48	200	8.99×10^{-2}	1.68×10^{-7}	14.2
80	81.93	200	4.46	8.34×10^{-8}	14.2
90	92.45	200	2.21	4.14	14.2
100	103.03	200	1.10×10^{-2}	2.05×10^{-8}	14.2

Values above this level are based on continuation of isothermal temperature, an arbitrary assumption resulting in P, ρ values progressively lower than more realistic temperature increase values.

Table 5.4-2: JPL MARS ATMOSPHERE, VM-3 (Continued)

Altitude Geopotential (km)	Altitude Geometric (km)	Temperature (°K)	Pressure (mb)	Density (gm/cm^3)	Density Scale Height (km)
150	156.93	200	3.28×10^{-4}	6.13×10^{-10}	14.2
200	212.52	200	9.81×10^{-6}	1.84×10^{-11}	14.2

$Po = 10.0$ mb

$\rho o = 1.36 \cdot 10^{-5}$ gm/cm^3

$To = 275$ °K

$T_{stratosphere} = 200$ °K

$M = 31.2$ (20% CO_2; 80% N_2 by volume)

$H_{stratosphere} = 19.3$ km

$go = 375$ cm/sec^2

Table 5.4-3: JPL MARS ATMOSPHERE, VM-7

Altitude Geopotential (km)	Altitude Geometric (km)	Temperature (°K)	Pressure (mb)	Density (gm/cm^3)	Density Scale Height (km)
0	0	275	5.00	6.82×10^{-6}	27.0
2	2.00	267	4.51	6.33	26.2
4	4.00	259	4.05	5.86	25.4
6	6.01	252	3.63	5.41	24.7
8	8.02	244	3.24	4.98	23.9
10	10.03	236	2.88	4.58	23.2
12	12.04	228	2.55	4.19	22.4
14	14.06	221	2.25	3.83	21.6
16	16.08	213	1.98	3.48	20.9
18	18.10	205	1.73	3.16	20.1
19.2	19.31	200	1.59	2.98	19.7
20	20.12	200	1.50	2.81	14.2
24	24.17	200	1.14	2.12	14.2
28	28.23	200	8.57×10^{-1}	1.60	14.2
32	32.30	200	6.47	1.21×10^{-6}	14.2
36	36.38	200	4.89	9.15×10^{-7}	14.2
40	40.48	200	3.69	6.91	14.2
44	44.58	200	2.79	5.22	14.2
48	48.69	200	2.11	3.94	14.2
50	50.75	200	1.83×10^{-1}	3.42	14.2
60	61.08	200	9.07×10^{-2}	1.70×10^{-7}	14.2
70	71.47	200	4.50	8.42×10^{-8}	14.2
80	81.93	200	2.23	4.17	14.2
90	92.45	200	1.10×10^{-2}	2.07	14.2
100	103.03	200	5.48×10^{-3}	1.02×10^{-8}	14.2

Values above this level are based on continuation of isothermal temperature, an arbitrary assumption resulting in P, ρ values progressively lower than more realistic temperature increase values.

Table 5.4-3: JPL MARS ATMOSPHERE, VM-7 (Continued)

Altitude Geopotential (km)	Altitude Geometric (km)	Temperature (°K)	Pressure (mb)	Density (gm/cm^3)	Density Scale Height (km)
150	156.93	200	1.64×10^{-4}	3.07×10^{-10}	14.2
200	212.52	200	4.90×10^{-6}	9.18×10^{-12}	14.2

$P_o = 5.0$ mb

$\rho_o = 6.82 \cdot 10^{-6}$ gm/cm^3

$T_o = 275$ °K

$T_{stratosphere} = 200$ °K

$M = 31.2$ (20% CO_2; 80% N_2 by volume)

$H_{stratosphere} = 19.3$ km

$g_o = 375$ cm/sec^2

Table 5.4-4: JPL MARS ATMOSPHERE, VM-8

Altitude Geopotential (km)	Altitude Geometrical (km)	Temperature (°K)	Pressure (mb)	Density (gm/cm^3)	Density Scale Height (km)
0	0	200	5.00	1.32×10^{-5}	13.8
2	2.00	189	4.08	1.14×10^{-5}	13.1
4	4.00	178	3.28	9.74×10^{-6}	12.3
6	6.01	168	2.61	8.24	11.6
8	8.02	157	2.04	6.90	10.8
10	10.03	146	1.57	5.70	10.1
12	12.04	135	1.19	4.64	9.36
14	14.06	124	8.74×10^{-1}	3.71	8.62
16	16.08	114	6.26	2.91	7.87
18	18.10	103	4.34	2.23	7.12
18.5	18.60	100	3.94	2.08	6.94
20	20.12	100	2.92	1.54×10^{-6}	5.05
24	24.17	100	1.32×10^{-1}	7.00×10^{-7}	5.05
28	28.23	100	6.01×10^{-2}	3.17	5.05
32	32.30	100	2.72	1.44×10^{-7}	5.05
36	36.38	100	1.23×10^{-2}	6.51×10^{-8}	5.05
40	40.48	100	5.59×10^{-3}	2.95	5.05
44	44.58	100	2.53	1.34×10^{-8}	5.05
48	48.69	100	1.15×10^{-3}	6.06×10^{-3}	5.05
50	50.75	100	7.73×10^{-4}	4.08×10^{-9}	5.05
60	61.08	100	1.07×10^{-4}	5.64×10^{-10}	5.05
70	71.48	100	1.48×10^{-5}	7.79×10^{-11}	5.05
80	81.93	100	2.04×10^{-6}	1.08×10^{-11}	5.05
90	92.45	100	2.82×10^{-7}	1.49×10^{-12}	5.05
100	103.03	100	3.90×10^{-8}	2.06×10^{-13}	5.05

Values above this level are based on continuation of isothermal temperature, an arbitrary assumption resulting in P, ρ values progressively lower than more realistic temperature increase values.

Table 5.4-4: JPL MARS ATMOSPHERE, VM-8 (Continued)

Altitude Geopotential (km)	Altitude Geometrical (km)	Temperature (°K)	Pressure (mb)	Density (gm/cm^3)	Density Scale Height (km)
150	156.93	100	1.97×10^{-12}	1.04×10^{-17}	5.05
200	212.52	100	9.93×10^{-17}	5.24×10^{-22}	5.05

$P_o = 5.0$ mb

$\rho_o = 1.32 \cdot 10^{-5}$ gm/cm^3

$T_o = 200$ °K

$T_{stratosphere} = 100$ °K

$M = 44$ (100% CO_2)

$H_{stratosphere} = 18.6$ km

$go = 375$ cm/sec^2

5.4.2 PLANET SOLAR RADIATION

The yearly mean black body effective temperature of Mars is assumed to be -13°C (9°F) for the sunlit side and -73°C (-100°F) for the dark side. Variations due to the eccentricity of the Martian orbit are accounted for by multiplying the yearly mean absolute temperature by the factor $\sqrt{1.0087 + 0.09405 \cos \Theta}$, where Θ is the heliocentric longitude of Mars from its perihelion. The solar intensity will vary as shown in Figure 5.3-1 during the orbital phase with ultraviolet--7.3%; visible--47.7%; and infrared--45%. Reflected solar radiation may be based on nominal Martian albedo of 0.15 with variations up to $\pm$10% possible.

5.4.3 CHARGED PARTICLE RADIATION

The galactic cosmic radiation will be as specified in Section 5.3.3.2. The solar wind, solar storms, and solar particle events environments are obtained by applying an inverse R-squared dependence of the flux to the flux specified in Section 5.3.3.1.

Areomagnetic trapped radiation is estimated conservatively at 10^{-5} that of the peak flux in the Earth's magnetosphere, that is:

Protons	($cm^{-2} sec^{-1}$)
2×10^3	E > 0.4 Mev
20	E > 4 Mev
1	E > 15 Mev
0.2	E > 34 Mev
0.04	E > 50 Mev
Electrons	
5×10^3	E > 0.5 Mev

The upper limit on the magnetosphere of Mars obtained by Mariner 4 removes the possibility of there being an intense trapped belt at Mars. Scaling of the Earth's belt by the energy density of the magnetic field would give an intensity of 10^{-6} that of the Earth's belt. In addition, the weaker magnetic field would require that the spectrum be quite soft. For a conservative estimate, the Mars trapped environment was defined as being 10^{-5} the peak intensity of the Earth's belt.

The effect of a weak Martian magnetic field on the incident solar protons would be to reduce their intensity. To maintain a conservative environment, this possible reduction has been neglected. This reduction would be most noted for the solar wind as it is most easily affected by a weak magnetic field, should such a field exist about Mars.

5.4.4 METEOROID

The same meteoroid flux equations used for Earth orbit was used for Mars orbit (see Section 5.2.6).

5.5 VENUS ENVIRONMENT

5.5.1 MODEL ATMOSPHERES

The Venus model atmospheres used in the IMISCD study are the NASA-MSFC model atmospheres published in TM-X-53142*. These atmospheres consist of an upper density model, a mean density model, and a lower density model and are based on the construction parameters shown in Table 5.5-1. The variation of temperature, pressure, and density with altitude is shown in Tables 5.5-2, 5.5-3 and 5.5-4, respectively, for the upper, mean, and lower density atmospheres. Figures 5.5-1 through 5.5-4 further illustrate important atmospheric parameters and compare the NASA-recommended mean Venus atmosphere data with the Earth's 1962 U.S. standard atmosphere.

The NASA mean density model was used for space vehicle design, and the upper and lower density models were used in determining the sensitivity of the space vehicle design to the uncertainty of the Venus atmosphere.

5.5.2 PLANET SOLAR RADIATION

The yearly mean black body effective temperature of Venus is assumed to be -39°C (-40°F).

The solar intensity will vary as shown in Figure 5.3-1 during the orbital phase with the following constituents:

Ultraviolet	7.3%
Visible	47.7%
Infrared	45.0%

During passage through the penumbra into total shadow, the variation of solar intensity may be approximated by a linear function of time. Reflected solar radiation at Venus may be based on a nominal Venusian a albedo of 0.76 with variations up to $\pm$10% possible.

5.5.3 CHARGED PARTICLE RADIATION

The galactic cosmic ray environment will be the same used for the interplanetary phase (Section 5.3.3.2) and for the solar particles employ the inverse R-squared dependence of the flux (Section 5.3.3.1). The trapped radiation about Venus is expected to be practically negligible. For a conservative estimate, use that given for the areomagnetically trapped radiation about Mars.

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*TM-X-53142, *Space Environment Criteria Guidelines for Use in Space Vehicle Development*, NASA Marshall Space Flight Center, September 1964

Table 5.5-1: CONSTRUCTION PARAMETERS—NASA-MSFC MODEL ATMOSPHERES

	NASA UDM Atmosphere	NASA MDM Atmosphere	NASA LDM Atmosphere
Surface Gravity (cm/sec^2)	832	886	914
Molecular Weight	29.6	32.0	40.0
CO_2 - N_2 (volume percent)	10-90	25-75	75-25
Radius (km)	6235	6045	5955
Surface Temperature (°K)	750	700	650
Surface P (Atm)	40	10	5
Surface ρ (grm/cc)	19.2×10^{-3}	5.57×10^{-3}	3.75×10^{-3}
Tropopause Height (km)	71	56	46
Troposphere Lapse Rate (°K/km)	-7.84	-8.49	-9.27
Stratosphere Temperature (°K)	194.2	225.2	224
Stratopause Height (km)	126	115	83
Thermosphere Lapse Rate (°K/km)	1.94	2.23	2.89

Table 5.5-2: NASA-MSFC UPPER DENSITY MODEL ATMOSPHERE

Altitude (km)	Temperature (°K)	Pressure (mb)	Density (grm/cm3)
0	750.0	4.05 E + 04	1.92 E + 02
5	712.6	3.31 E + 04	1.65 E - 02
10	674.8	2.67 E + 04	1.41 E - 02
20	598.3	1.68 E + 04	1.00 E - 02
30	520.7	9.94 E + 03	6.80 E - 03
40	442.1	5.40 E + 03	4.35 E - 03
50	362.7	2.61 E + 03	2.65 E - 03
75	194.2	1.50 E + 02	2.75 E - 04
100	194.2	3.67 E + 00	6.73 E - 06
150	242.6	3.56 E - 03	5.22 E - 09
200	339.4	2.75 E - 05	2.88 E - 11
300	533.0	4.60 E - 08	3.07 E - 14
400	726.6	6.53 E - 10	3.20 E - 16

Note: 3.56 E - 03 means 3.56×10^{-3}

Table 5.5-3: NASA-MSFC MEAN DENSITY MODEL ATMOSPHERE

Altitude (km)	Temperature (°K)	Pressure (mb)	Density (grm/cm^3)
0	700.0	1.01 E + 04	5.57 E - 03
5	659.7	7.88 E + 03	4.60 E - 03
10	619.0	6.04 E + 03	3.76 E - 03
20	536.4	3.36 E + 03	2.41 E - 03
30	452.1	1.69 E + 03	1.44 E - 03
40	366.1	7.40 E + 02	7.78 E - 04
50	278.6	2.59 E + 02	3.59 E - 04
75	225.2	6.98 E + 00	1.19 E - 05
100	225.2	1.77 E - 01	3.02 E - 07
150	305.6	2.63 E - 04	3.31 E - 10
200	417.3	2.94 E - 06	2.71 E - 12
300	640.5	6.99 E - 09	4.20 E - 15
400	863.8	1.18 E - 10	5.25 E - 17

Table 5.5-4: NASA-MSFC LOWER DENSITY MODEL ATMOSPHERE

Altitude (km)	Temperature (°K)	Pressure (mb)	Density (grm/cm^3)
0	650.0	5.07 E + 03	3.75 E - 03
5	608.0	3.57 E + 03	2.83 E - 03
10	565.2	2.46 E + 03	2.09 E - 03
20	477.1	1.06 E + 03	1.07 E - 03
30	384.7	3.84 E + 02	4.81 E - 04
40	286.6	1.05 E + 02	1.76 E - 04
50	224.0	1.74 E + 01	3.73 E - 07
75	224.0	1.42 E - 01	3.05 E - 07
100	273.0	1.65 E - 03	2.90 E - 09
150	417.4	3.33 E - 06	3.84 E - 12
200	561.7	4.66 E - 08	3.99 E - 14
300	850.3	1.38 E - 10	7.78 E - 17
400	1139.0	2.59 E - 12	1.09 E - 18

5.5.4 METEOROID

The same meteoroid flux equations used for Earth orbit will be used for Venus orbit (see Section 5.2.6).

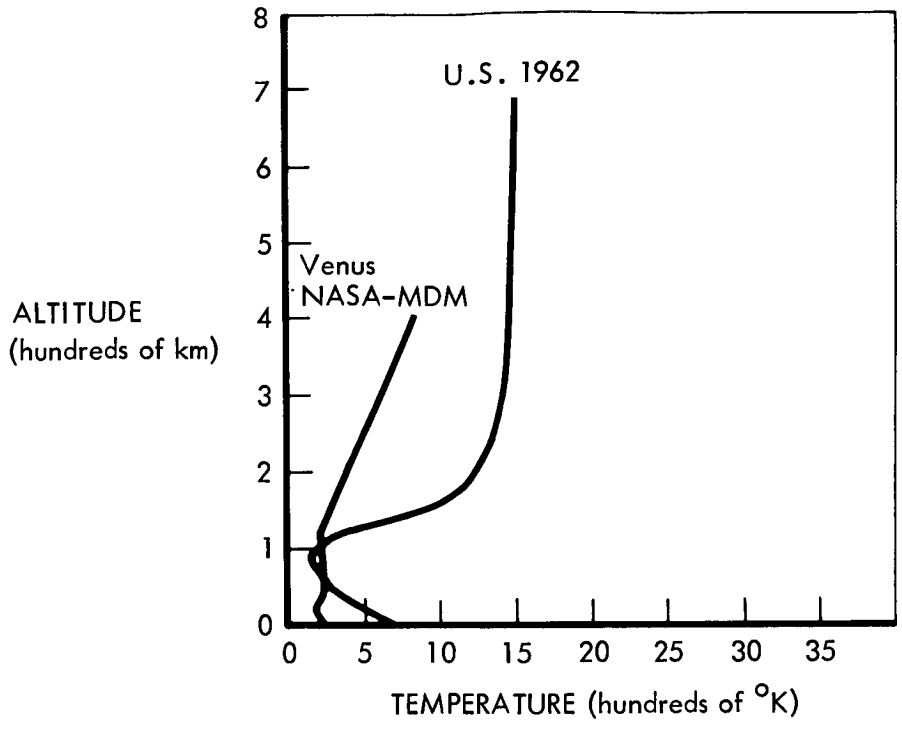


Figure 5.5-1: TEMPERATURE VERSUS ALTITUDE — VENUS AND EARTH

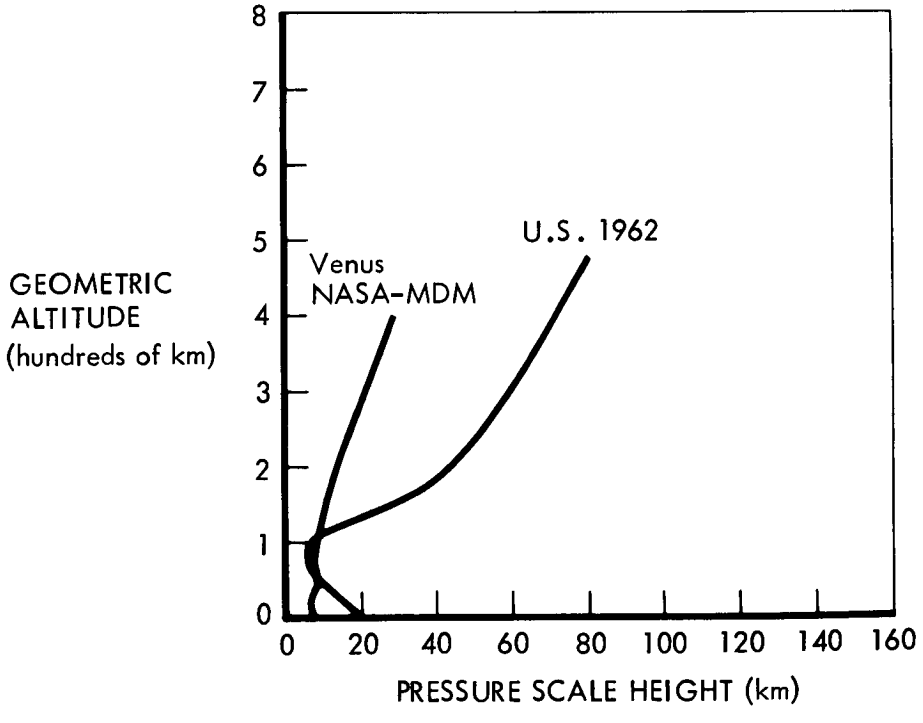


Figure 5.5-2: PRESSURE SCALE HEIGHT VERSUS ALTITUDE — VENUS AND EARTH

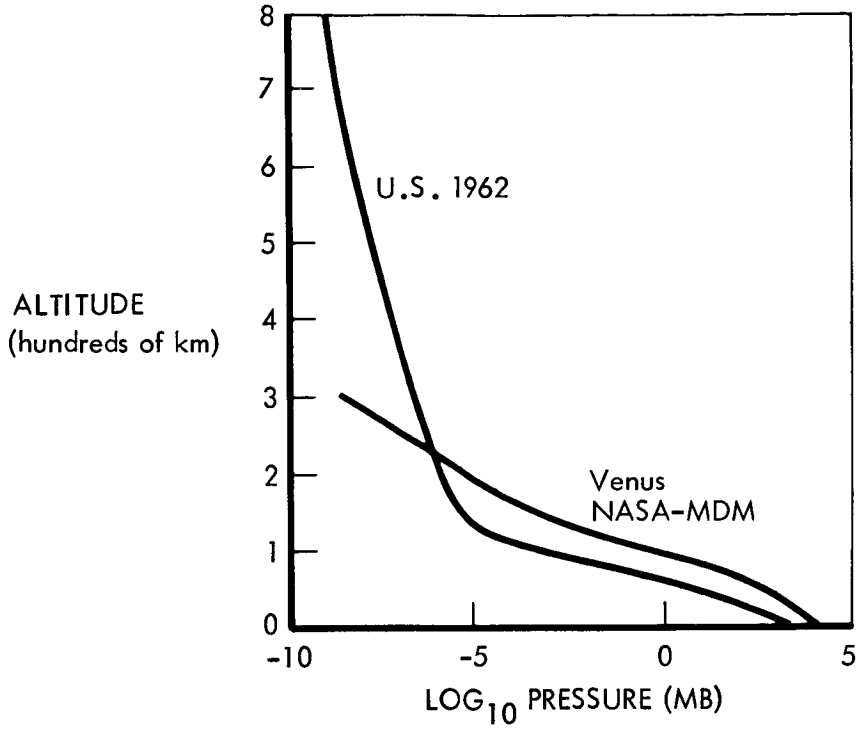


Figure 5.5-3: PRESSURE VERSUS ALTITUDE — VENUS AND EARTH

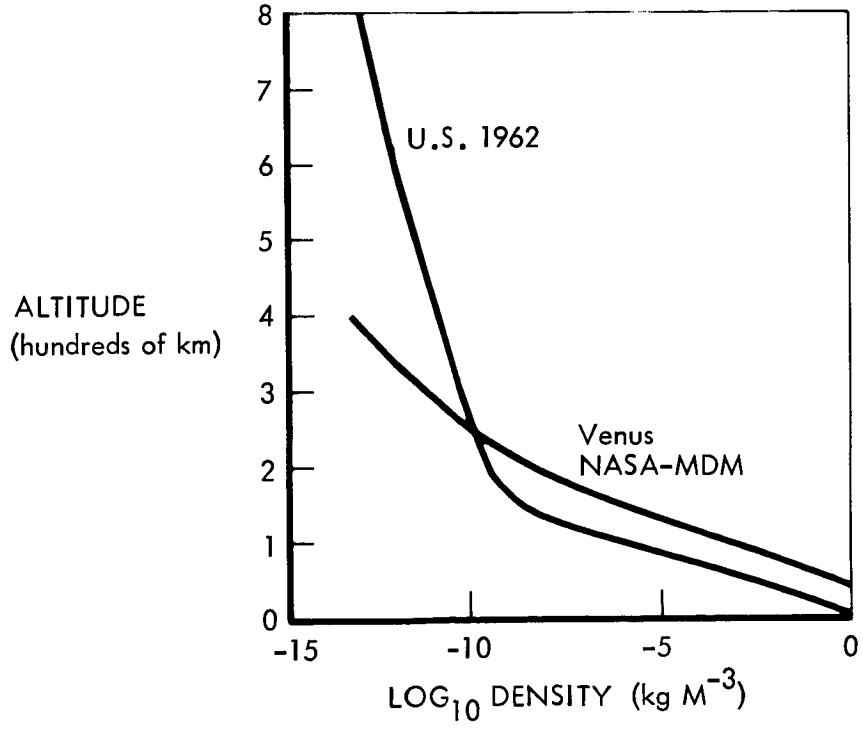


Figure 5.5-4: DENSITY VERSUS ALTITUDE — VENUS AND EARTH